Solar Orbiter

Payload Definition Document

(PDD)

Reference	SCI-A/TA
Issue	1
Revision	7
Date of issue	19 February 2003

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Issue/ Rev.	Section / Page	Date	Changes
1 / 0	All	19/12/2002	
1 / 1	3.4.1	20/12/2002	New writing of Scientific drivers
1 / 1	MAG data sh.	20/12/2002	Peak data rate increased
1 / 1	MAG data sh.	20/12/2002	Min data rate decreased
1 / 1	MAG data sh.	20/12/2002	TRL, DML changed
1 / 2	5.2	6/1/2003	HI write up
1 / 2	HI data sh.	6/1/2003	Objectives, pointing, peak power, stand-by power, heater power, DML (optics)
1 / 2	4.5	6/1/2003	RAD description updated
1 / 2	RAD data sh.	6/1/2003	Objectives, description, heritage, FOV, S/C requirements, mass, dimensions, power, stand-by, telemetry, peak rate, data volume, heat load, thermal drift, cleanliness, mechanisms, FOV, model philosophy, DML
1 / 2	4.3	8/1/2003	EUI description updated
1 / 2	EUI data sh.	8/1/2003	Objectives, spectral range, Operating T, FOV, S/C requirements, mass, mechanisms, unobstructed FOV, critical items
1 / 3	5.1	15/1/2003	STIX description updated
1 / 3	STIX data sh.	15/1/2003	FOV, contingency (1 kg), dimensions
1 / 3	DUD data sh.	15/1/2003	DUD data sheet included
1 / 4	4.5.4	31/1/2003	RAD power lowered
1 / 4	RAD data sh.	31/1/2003	Average power lowered
1 / 5	4.1.4	6/2/2003	Table 4.2 updated including margin
1 / 5	4.1.4	6/2/2003	Table 4.3: margin added, focusing mechanism removed?
1 / 6	All	14/2/2003	Tables and figures consistency check
1 / 6	3.3	14/2/2003	Add missing informations for CRS
1 / 6	Table 1.2	14/2/2003	Filled in missing informations
1 / 6	Table 2.1	14/2/2003	Consistency check
1 / 7	Table 2.1	19/2/2003	Operating temperature range checked

1. INTRODUCTION

The ESA Solar Orbiter mission consists of a single spacecraft.

This document is a compilation of the Solar Orbiter model payload requirements. Information has been provided by selected experts and the inputs have been reviewed and edited by the Advanced Concept and Science Payload Office of ESA.

The principal features of the model payload are summarized in Chapter 2. The requirements include some margin and are provided with varying degree of confidence, depending on the possible experience inherited from earlier missions.

A Contact Person that will be able to answer questions related to the individual instruments has been nominated for each instrument. Requests to the contact persons shall be done in coordination with ESA. The mail and electronic addresses, telephone and fax numbers of the Contact Persons are given in Table 1-2.

Table 1-1: Solar Orbiter model payload instruments

Instrument	Acronym	Contact Person
Solar Wind Plasma Analyzer	SWA	McComas
Radio and Plasma Wave Analyzer	RPW	S.Bale
Radio Sounding	CRS	Passive instrument
Magnetometer	MAG	C.Carr
Energetic Particle Detector	EPD	H.Kunow
Dust Detector	DUD	I.Mann
Neutral Particle Detector	NPD	Hilchenbach
Neutron Detector	NED	Barraclough
Visible Imager and Magnetograph	VIM	Valentin Martinez
EUV Imager and Spectrometer	EUS	R.Harrison
EUV Imager	EUI	J.-M.Defise
UV Coronograph	UVC	Fineschi
Radiometer	RAD	I.Ruedi
Spectrometer/Telescope Imaging X-rays	STIX	Hurford
Heliospheric Imager	HI	Korendyke
Gamma-ray detector	GRD	R.Lin

Table 1-2: Contact Persons

Name	Address	Telephone	Fax	E-mail
McComas	Southwest Research Institute, P.O. Drawer 28510, San Antonio, TX 78228-0510	+1-210-522-5983	+1-210-520-9935	dmccomas@swri.edu
S.Bale	Space Sciences Laboratory, University of California, Berkeley, CA 94720-7450	+1-510-643-3324	+1-510-643-8302	bale@ssl.berkeley.edu
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J.-M.Defise	Centre Spatial de Liège, Université de Liège, Parc Scientifique du Sart Tilman, Avenue du Pré-Aily, B-4031, Angleur-Liège, Belgium	+32-43-676668	+32-43-675613	jmdefise@ulg.ac.be
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I.Mann	Institut fuer Planetologie, Westfaelische Wilhelms – Universitaet, Wilhelm - Klemm - Str. 10, 48149 Muenster, Germany	+49-251-83-39081	+49-251-83-36301	imann@uni-muenster.de
Hilchenbach	Max-Planck-Institut für Aeronomie, Max-Planck-Straße 2, 37191 Katlenburg-Lindau, Germany	+49-5556-979-162	+49-5556-979-240	hilchenbach@linmpi.mpg.de
Barraclough	Los Alamos National Laboratory, P.O. Box 1663 Los Alamos, NM 87545	+1-505-667-8244	+1-505-665-7395	bbarraclough@lanl.gov
Valentin Martinez	Instituto de Astrofisica de Canarias, c/ Via Lactea s/n, La Laguna, 38200, Tenerife, Spain	+34-922-605237	+34-922-605210	vmp@ll.iac.es
R.Harrison	Rutherford Appleton Laboratory, Chilton Didcot, Oxfordshire, OX11 0QX, UK	+44-1235-446884	+44-1235-445848	R.A.Harrison@rl.ac.uk
S.Fineschi	Osservatorio Astronomico di Torino, 20 Strada Osservatorio, 10025 Pino Torinese, Italy	+39-011-810-1919	+39-011-810-1930	fineschi@to.astro.it
I.Ruedi	WRC/PMOD, Dorfstrasse 33, CH-7260 Davos Dorf, Switzerland	+41-81-417-5133	+41-81-417-5100	i.ruedi@pmodwrc.ch
Hurford	Space Sciences Laboratory, University of California, Berkeley, CA 94720-7450	+1-510-643-9653		ghurford@ssl.berkeley.edu
Korendyke	Naval Research Laboratory Code 7662K, Washington, DC 20375	+1-202-767-3144	+1-202-767-5636	ckorendyke@ssd5.nrl.navy.mil
R.Lin	Space Sciences Laboratory, University of California, Berkeley, CA 94720-7450	+1-510-642 1149		rlin@ssl.berkeley.edu

2. REQUIREMENT SUMMARY

The major instruments requirements are summarised in Table 2-1 respective.

Table 2-1: Solar Orbiter Instrument Requirements Summary

				DATA	POINTING			FIELD OF	TEMPERATURE		MATURITY	SPECIAL
	MASS	POWER	VOLUME	RATE	DIR.	ACC.	KNOWL.	VIEW	OPERATING	NON-OP	LEVEL	REQUESTS
	Kg (2)	W(av.)	L	kbps(av.)		arcsec	arcsec	deg	deg C	deg C	(1)	
Heliospheric in-situ instruments												
SWA	15.5	12.5	70	14	Sun	1800	1800	60	-30/+40	-35/+45	2	
RPW	11.3	7.5	N/A	5	N/A	N/A	N/A	4□	TBD	TBD	2	Several units
CRS	0.2	3	0.13	0.01	N/A	N/A	N/A	N/A	TBD	TBD	1	Stabilized quartz clock
MAG	0.88	1.5	1.51	0.8	N/A	N/A	N/A	N/A	-30/+60	-100/+100	2	
EPD	8.68	10.2	51.7	2.8	Various	N/A	N/A	Various	-30/+30	TBD	2	7 units
DUD	1	0.6	1	4	N/A	N/A	N/A	N/A	-30/+40	TBD	2	
NPD	1.3	2	4	0.32	7.5 deg off Sun	360	N/A	8 x 8	-30/+50	TBD	2	
NED	3.5	3	3.38	0.16	Nadir	N/A	N/A	4□	-20/+30	TBD	2	Twin of GRD
Remote sensing instruments												
VIM	27	23	140	20	Sun	<1	TBD	2.7 x 2.7	-5/+200	TBD	5	Cool detectors Hot optics
EUS	23	25	200	17	Sun	TBD	TBD	0.6 x 0.6	-80/100	TBD	3	Cool detectors Hot optics
EUI	42	25	327	20	Sun	TBD	60	5 x 5 0.1 x 0.1	-80/100	TBD	3	Cool detectors Hot optics
UVC	23	25	144	72	Sun	N/A	N/A	7 x 7	+20/+30	TBD	3	
RAD	6.7	7.4	6.92	0.009	Sun	N/A	N/A	9 x 9	+10/+50	TBD	3	
Additional instruments												
STIX	4	4	21.6	0.2	Sun	TBD	TBD	3 x 3 0.5 x 0.5	-25/+25	TBD	3	Roll requirements
HI	9.6	2	67.5	1.6	Sun	<60	TBD	>30	TBD	TBD	2	Two instruments
GRD	3.5	3	3.38	0.16	Nadir	N/A	N/A	4□	-20/+30	TBD	2	Twin of NED

(1) Maturity levels:
1 Existing hardware
2 Existing +minor modifications
3 Existing +major modifications
4 New, Detailed design level
5 New, Preliminary design level
6 Concept only

(2) W/O contingency

Status 19-2-2003

3. Heliospheric instruments

3.1 Solar Wind Plasma Analyzer (SWA)

3.1.1 INSTRUMENT DESCRIPTION

SCIENTIFIC DRIVERS

Three principal science goals require the inclusion of a solar wind plasma analyser in the payload:

- to provide observational constraints on kinetic plasma properties for a fundamental and detailed theoretical treatment of all aspects of coronal heating;
- to investigate charge- and mass-dependent fractionation processes of the solar wind acceleration process in the inner corona;
- to correlate comprehensive in-situ plasma analysis and compositional tracer diagnostics with space-based and ground-based optical observations of individual stream elements.

Furthermore, the SWA will

- investigate in detail ^3He and "unusual" charge states in CME-related flows and to correlate the observations with other in-situ particle and field data and with optical observations;
- investigate the "recycling" of solar wind ions on dust grains in the distance range which has been located as the "inner source". Freshly produced pick-up ions from this inner source are specially suited as test particles for studying the dynamics of incorporation of these particles into the solar wind.

The SWA will measure separately the three-dimensional distribution functions of the major solar wind constituents: protons, α-particles and electrons. The basic moments of the distributions, such as density, velocity, temperature tensor, and heat flux vector will be obtained under all solar wind conditions and be sampled sufficiently rapidly to characterise fully the fluid and kinetic state of the wind. In this way we will be able to determine possible non-gyrotropic features of the distributions, ion beams, temperature anisotropies, and particle signatures of wave excitation and dissipation.

In addition, measurements of representative high-FIP elements (the C, O, N group) and of low-FIP elements (such as Fe, Si or Mg) will be carried out in order

- to obtain their abundances, velocities, temperature anisotropies and charge states;
- to probe the wave-particle couplings (heavy-ion wave surfing);
- determine the freeze-in temperatures (as a proxy for the coronal electron temperature).

INSTRUMENT CONCEPT

In view of the limited resources of mass, volume and telemetry allocated to the SWA, a compromise between sensitivity, mass/charge- and mass- and time resolution has to be found. The SWA has to cover a large dynamic range in ion fluxes. Since there is an enormous difference between the proton fluxes at perihelion (typically 10^{14} m^{-2} s^{-1}) and the fluxes of relevant minor ion tracers at 1 AU (e.g. Fe^{10+} at typically 10^8 m^{-2} s^{-1} etc.) it is suggested to implement three different sensors:

1. A Proton/α-particle Sensor (PAS) with the principal aim to investigate the velocity distribution of the major ionic species at a time resolution equivalent to the ambient proton cyclotron frequency.
2. An Electron Analysers System (EAS) consisting of two or three combined sensor heads covering a solid angle of almost 4π and allowing the determination of the principal moments of the electron velocity distribution at high time resolution.
3. A Heavy Ion Sensor (MIS) which allows the independent determination of the major charge states of oxygen and iron and a coarse mapping of the three-dimensional velocity distribution of some prominent minor species. Also, pick-up ions of various origins, such as weakly-ionised species (C^+, N^+, Ne^+, Mg^+, Si^+, etc.), will be measured.

The PAS is designed to resolve the relevant ion-kinetic scales. It will achieve a time resolution which is compatible with the characteristic ion cyclotron periods near perihelion. The characteristic time scale is about 10 ms. For the minor species (heavier than He) a time resolution of 1 s is needed.

To fulfil the requirements of compositional diagnostics, at least the charge state distributions of oxygen and iron should be measured routinely and with a time resolution of better than 10 s.

The instrument specifications are outlined in Table 3.1:

Sensor	Particles	Field of View	Energy resolution	Angular resolution	Time resolution
PAS	Protons and α-particles	Conical ±45°	E/ΔE = 10	Azimuth 30° Zonal ±5°	10 ms
EAS	Electrons	4π	E/ΔE = 10	Azimuth 30° Zonal ±5°	10 ms
HIS	O, (3He, C, Ne, Mg, Si), Fe	Conical ±45°			1 s

Table 3.1: Specifications of the SWA sensors.

The instrument package proposed for SWA is a further development of instruments successfully flown on several missions in the past, e.g., SWICS on Ulysses, CELIAS/CTOF on SOHO, TRIPLET on INTERBALL. A possible scheme for PAS based upon the design of TRIPLET is shown in Figure 3.1.

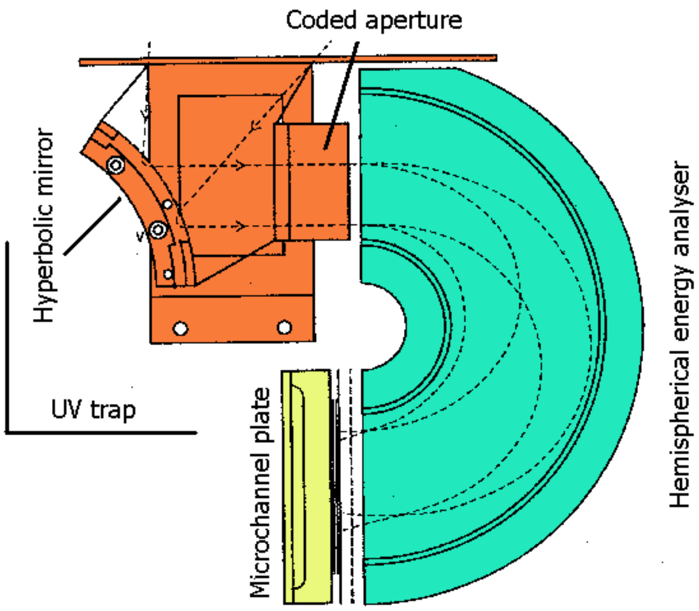

Figure 3.1: Schematics of PAS (following the design of TRIPLET on INTERBALL). A hyperbolic mirror maps the angular velocity distribution of solar wind ions onto a coded aperture. With this measure the information on the position at the entrance of the hemispherical energy analyser is transmitted (at the cost of ~ 50% reduction in beam intensity) to the exit aperture of the analyser. The additional information on energy per charge is contained in the exit location and convoluted with the entrance location. Deconvolution of the entrance angle and the energy per charge by the onboard DPU allows the reconstruction of the complete three-dimensional velocity distribution of solar wind protons and α particles with a time resolution of 10 ms.

3.1.2 Orbit, Operations and Pointing Requirements

3.1.3 Accommodation

3.1.4 Interface and Physical Resource Requirements

Telemetry

The transmission of velocity distributions of the major species (protons, electrons and α particles) drives the telemetry requirements of SWA. In the normal mode of operation this information should be compressed to transmit the principal moments of the distributions. Periodically however, the transmission of the complete velocity distributions of protons and α particles with a (velocity) resolution of 10 x 10 x 10 pixels at a depth of 5 bits is needed. This requires 5 kb per spectrum. Correspondingly, at a time resolution of 10 ms, the required telemetry rate is 500 kb/s. This rate should be maintained during 1s, i.e. for some 100 ion cyclotron periods to cover a typical growth time of plasma instabilities. With an average telemetry rate of 5 kb/s a duty cycle of the order of 1% can be achieved.

Allocations of mass, volume, and power.

- A mass allocation for HIS of 8 kg, 3 kg for PAS and 4.5 kg for EAS (excluding the electronics) is foreseen.
- The volume allocation for HIS is 40 x 40 x 30 cm^3, and 40 x 30 x 20 cm^3 for PAS and 7 x 7 x 7 cm^3 for each of the two EAS sensors.
- The total power allocated to SWA is 11 W.

Thermal and radiation requirements

The instrument should be able to operate under the specified spacecraft conditions. Special attention has to be devoted to the extreme solar UV radiation flux. The design of the particle instruments should reject this radiation to a sufficiently low level in order not to compromise the detection of rare particles.

Electrostatic cleanliness

Special care has to be taken to ensure near-equipotential conditions in the vicinity of the EAS entrance apertures.

Instrument summary

Instrument	**Mass (kg)**	**Power (W)**	**Volume (cm^3)**	**Data rate (b/s)**
PAS	3	2	24000	2000
EAS	4.5	2	686	TBD
HIS	8	7	48000	10000
Total SWA	15.5	11	72686	>12000

Table 3.2: SWA resource requirements.

3.1.5 Cleanliness, Ground Operations and Other Requirements

Special care has to be taken to ensure near-equipotential conditions in the vicinity of the EAS entrance apertures.

3.1.6 Open Points and Critical Issues

SWA Instrument data sheet (A compilation of EAS, PAS, and HIS sensor inputs)

Name / acronym	**Solar Wind Analyzer (SWA) – a combination of Electron Analyzer System (EAS), Proton/□-particle Sensor (PAS), and Heavy Ion Sensor (HIS-previously called MIS).**

Objectives	1) to measure core and halo electrons in the solar wind with high time resolution and 4pi coverage (EAS).
	2) to detect electron dynamics and kinetic evolution in the quiet solar wind and during dynamic events such as shocks and high speed streams (EAS).
	3) to measure the solar wind proton and alpha particle beams with high time and angular resolution (PAS).
	4) to investigate the evolution of the velocity distributions of the major ions with a time resolution equivalent to the ambient proton cyclotron frequency (PAS).
	5) to measure the key solar wind elemental and ionic composition and relate the compositional signatures to the solar wind origin (HIS).
	6) to investigate the evolution of the velocity distributions of the heavy ions and their relation to the ambient plasma properties, such as the dynamic state of the solar wind turbulence (HIS).
	7) to provide bulk ion moments for the solar wind heavy ion distributions over the full range of heliocentric distances observed (HIS).

General description	The Solar Wind Analyzer (SWA) – a combination of three sensors: the Electron Analyzer System (EAS), the Proton/□-particle Sensor (PAS), and the Heavy Ion Sensor (HIS). The three sensors share a common DPU (not included in the resources given here) or could be supported be an in situ instrument payload DPU. All three sensors utilize electrostatic analysis; This technique is implemented in different ways for each sensor and combined with other analysis techniques (e.g., electrostatic deflection, energy/angle imaging, and time of flight). Additional detail for each sensor is broken out in the three separate instrument data sheets attached below.

Reference P/L and/or heritage	Heritage is drawn from a variety of flight proven sensor designs including Cluster PEACE, Cassini CAPS ELS, ACE SWEPAM, Ulysses/SWOOPS, ACE and Ulysses SWICS, and SOHO CTOF.

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Electrostatic analyzers with a combination of electrostatic deflection, time-of-flight analysis, and position sensing	
Operating T	K	Min 243°; Max 313°	-30 to 40° C
Optics / antennas			
Type	N/A	Electro-optics	
FOV	Rad	Most of 4p sr (EAS)	Covered with 3 heads
Clear FOV	deg	60 deg cone centered on Sun (PAS and HIS)	Required from S/C, actual FOV smaller
Energy passband	E/q	200 eV/q - 8 keV/q (PAS)	
Energy passband	E/q	200 eV/q - 40 keV/q (HIS)	
Pointing	N/A	Sun pointing	0.5° stability and knowledge

Configuration			
Physical Units	No	5: Single mechanical structure for each of PAS and HIS and 3 EASs	
Layout	N/A	3 -nominal 360 degree fans mutually perpendicular (EAS)	One FOV in ecliptic plane to include Parker spiral angle
		Several canted fans crossing beam (PAS and HIS)	Could be conical FOV with energy/angle imaging
Location on S/C	N/A	Sides (2) Third analyzer for core electrons in shadow (EAS); Sun-facing side (PAS & HIS)	
Physical			
Mass	kg	15.5: 4.5 (EAS), 3 (PAS), and 8 (HIS)	Does not include DPU; does include FPGA based integration algorithms for data management for HIS only.
Volume	cm3	70k total: 700 (EAS), 24k (PAS), 48k (HIS)	
Power			
Average	W	11: 2(EAS), 2(PAS), 7(HIS)	
Peak power	W	12.5: 2(EAS), 2.5(PAS), 8(HIS)	
Stand-by	W	8.5: 2(EAS), 1.5(PAS), 5(HIS)	
Data rate / volume			
Data rate	bit/sec	14k: 2k (EAS), 2k(PAS), 10k (HIS)	
Data volume /orbit	KByte	See DPU input	
Own data storage	MByte	To be done in DPU	
Thermal			
Heat load to radiator	W	Depends on location	dissipated to S/C mounting plate
Operating T	K	Min 243°; Max 313°	-30 to +40° C
Survival T	K	Min 238°; Max 318°	-35 to +45° C
Cleanliness			
EMC requirements	N/A	Normal S/C, not sensitive	
DC magnetic	N/A		
Particulate	N/A	Class 10,000 clean room at all times	
water, hydrocarbon, etc	N/A	MCPs very sensitive	Purge through testing and in fairing highly desirable
Miscellaneous			
Mechanisms	No.	Open once door covers	Pyro or bimetal motors
Alignment		0.5° total stability, knowledge	
Orbit requirements		SO orbit is ideal	
AIT/AIV requirements		N/A	

Development approach / schedule

Preferred model philosophy	e.g. EBB, EM, QM, FM + FS (EAS)
	e.g. PT, EM/QM, FM + FS (PAS, HIS)
Estimated development time	EBB (1 yr), EM (1 yr), QM (2 yr), FM (1 yr), FS (1 yr) (EAS)
	PT (1 yr), EM/QM (2 yr), FM (1 yr), FS (1 yr) (PAS, HIS)

Areas considered as critical.

Critical area /unit/ subsystem	**Remarks, proposed risk-mitigating measures.**
Electro-optics	Simultaneous energy/angle imaging design and electrostatic sweeping optics needs development
AMP/TOF chip	Highly desirable to meet mass and power requirements

SSD electronics	Highly preferable to develop integrated readout electronics for SSD pixels.

Technology readiness – Design maturity level

Unit/ subsystem	TRL	DML	Justification and remarks

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

SWA-EAS Instrument data sheet

Name / acronym	**Electron Analyzer System/EAS**

Objectives	1) to measure core and halo electrons in the solar wind with high time resolution and 4pi coverage
	2) to detect electron dynamics and kinetic evolution in the quiet solar wind and during dynamic events such as shocks and high speed streams
	3) to provide electron density, pressure tensor, heat flux, pitch angle distribution and 3d velocity distribution functions

General description	System of electrostatic analyzers with electrostatic deflection

Reference P/L and/or heritage	Modification from Cluster PEACE and Cassini CAPS ELS taking Ulysses data into account and taking account of orbit. For example geometric factor can be lower, but particular care must be taken with suppression of UV, thermal design, and higher spacecraft potential. Separate sensor may be needed for core electrons.
	Cluster PEACE, Cassini CAPS ELS
	Ulysses SWEPAM

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Electrostatic analyzers with electrostatic deflection	
Spectral range	Nm/eV	0-5 keV	
Operating T	K	Max 313	
Optics / antennas			
Type	N/A	N/A	
FOV	Rad		
Bandpass	Nm/eV		
Pointing	N/A		
Configuration			
Physical Units	No	3	
Layout	N/A	Nominal 360 degree fans mutually perpendicular	One FOV in ecliptic plane to include Parker spiral angle
Location S/C	N/A	Sides (2) Third analyzer for core electrons in shadow	
Physical			
Mass, total	kg	3	
Mass unit 1	kg	0.75	
Mass unit 2	kg	0.75	Unit 3 0.75kg also
Dimension 1	cm	7 x 7 x 7	
Dimension 2	cm	7 x 7 x 7	
Power			
Average	W	2	
Peak power	W		
Stand-by	W		
Data rate / volume			
Average data rate	Bits/sec		
Peak data rate	Bits/sec		
Data volume /orbit	KByte		
Own data storage	MByte		

Thermal			
Heat load to radiator	W	e.g. dissipated to SVM I/F plate	
Operating T range	K		
Other requirements	N/A	e.g. Peltier element	
Cleanliness			
EMC requirements	N/A		
DC magnetic	N/A		
Particulate	N/A		
Miscellaneous			
Mechanisms	No.	No. and type, shutter, filters, etc.	
Alignment		Wrt S/C, other instruments	
Orbit requirements		e.g. altitude, inclination	
AIT/AIV requirements		e.g. rad. sources, biological issues	

Development approach / schedule

Preferred model philosophy	e.g. EBB, EM, QM, FM + FS
Estimated development time	EBB (1 yr), EM (1 yr), QM (2 yr), FM (1 yr), FS (1 yr).

SWA-PAS - Instrument data sheet

Name / acronym	**Proton/□-particle Sensor (PAS) – part of SWA (Solar Wind Plasma Analyzer Instrument)**

Objectives	1) to measure the solar wind proton and alpha particle beams with high time and angular resolution
	2) to investigate the evolution of the velocity distributions of the major ions with a time resolution equivalent to the ambient proton cyclotron frequency.
	3) to provide bulk ion moments for the solar over the full range of heliocentric distances observed.

General description	Energy per charge analysis with high sensitivity to achieve required time resolution. May be based on a system of electrostatic analyzers with electrostatic deflection and/or energy/angle imaging.

Reference P/L and/or heritage	Design evolved from numerous E/q analyzers but taking account of orbit and needed high time resolution. Particular care must be taken with suppression of UV and thermal design. Multiple smaller analyzers or simultaneous energy/angle imaging is required.
	ACE SWEPAM, Ulysses/SWOOPS
	CASSINI IMS, possibly INTERBALL TRIPLET

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Electrostatic analyzers	
Spectral range			
Operating T	K	Min 243°; Max 313°	-30 to 40° C
Optics			
Type	N/A	Electro-optics	
Clear FOV	deg	60 deg cone centered on Sun	Required from S/C, actual FOV smaller
Energy passband	E/q	200 eV/q - 8 keV/q	
Pointing	N/A	Sun pointing	0.5° stability and knowledge
Configuration			
Physical Units	No	Single mechanical structure	Multiple apertures
Layout	N/A	Several canted fans crossing beam	Could be conical FOV with energy/angle imaging
Location S/C	N/A	Sun-facing side	
Physical			
Mass, total	kg	3	Does not include DPU
Dimensions	cm	40x30x20	
Power			
Average	W	2	
Peak power	W	2.5	
Stand-by	W	1.5	
Data rate / volume			
Average data rate	bit/sec	2k	
Peak data rate	bit/sec	2k	
Data volume /orbit	KByte	See SWA DPU input	
Own data storage	MByte	To be done in SWA DPU	
Thermal			
Heat load to radiator	W	Depends on location	dissipated to S/C mounting plate
Operating T	K	Min 243°; Max 313°	-30 to +40° C

Survival T	K	Min 238°; Max 318°	-35 to +45° C
Cleanliness			
EMC requirements	N/A	Normal S/C, not sensitive	
DC magnetic	N/A		
Particulate	N/A	Class 10,000 clean room at all times	
water, hydrocarbon, etc	N/A	MCPs very sensitive	Purge through testing and in fairing highly desirable
Miscellaneous			
Mechanisms	No.	Open once door covers	Pyro or bimetal motors
Alignment		0.5° total stability, knowledge	
Orbit requirements		SO orbit is ideal	
AIT/AIV requirements		N/A	

Development approach / schedule

Preferred model philosophy	e.g. PT, EM/QM, FM + FS
Estimated development time	PT (1 yr), EM/QM (2 yr), FM (1 yr), FS (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	**Remarks, proposed risk-mitigating measures.**
Electro-optics	Simultaneous energy/angle imaging design would need development
AMP/TOF chip	Highly desirable to meet mass and power requirements

SWA-HIS - Instrument data sheet

Name / acronym	**Heavy Ion Sensor[1] – part of SWA (Solar Wind Plasma Analyzer Instrument) Note – previously called MIS (Minor Ion Sensor)**

Objectives	1) to measure the key solar wind elemental and ionic composition and relate the compositional signatures to the solar wind origin.
	2) to investigate the evolution of the velocity distributions of the heavy ions and their relation to the ambient plasma properties, such as the dynamic state of the solar wind turbulence.
	3) to provide bulk ion moments for the solar wind heavy ion distributions over the full range of heliocentric distances observed.

General description	Energy per charge analysis, followed by linear time-of-flight and total energy analysis. Use position-sensitive mapping to deduce velocity vector direction. Also, consider electrostatic deflection or advanced collimator designs to adjust sensitivity to within constraints of electronics throughput. Perhaps multiple apertures.

Reference P/L and/or heritage	Design evolved from Solar Wind Ion Composition experiment, which is currently the only comparable composition instrument in use. Particular care must be taken with suppression of UV and thermal design. Also, dynamic range considerations are likely design drivers for both, electronic and deflection design.
	ACE and Ulysses SWICS.
	SOHO CTOF.

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Electrostatic analyzer. Time-of-flight telescope with position sensing capability. Low-threshold Solid State Detectors, pixilated, with parallel readout paths.	
Spectral range			
Operating T	K	Min 243°; Max 313°	-30 to 40° C
Optics			
Type	N/A	Electro-optics deflection system	
Clear FOV	deg	60 deg cone centered on Sun	Required from S/C, actual FOV smaller
Energy passband	E/q	200 eV/q - 40 keV/q	
Pointing	N/A	Sun pointing	0.5° stability and knowledge
Configuration			
Physical Units	No	Single mechanical structure	Consider multiple apertures.
Layout	N/A	Canted fan crossing beam with electrostatic manipulation capability.	Could be conical FOV with energy/angle imaging
Location S/C	N/A	Sun-facing side	
Physical			
Mass, total	kg	8	Does not include DPU, but does include FPGA based integration algorithms for data management.
Dimensions	cm	40x40x30	
Power			

Average	W	7	
Peak power	W	8	
Stand-by	W	5	With detector voltages on.
Data rate / volume			
Average data rate	bit/sec	10k	Assumes compression in digital electronics of sensor.
Peak data rate	bit/sec	10k	
Data volume /orbit	KByte	See SWA DPU input	
Own data storage	MByte	To be done in SWA DPU	
Thermal			
Heat load to radiator	W	Depends on location	dissipated to S/C mounting plate
Operating T	K	Min 243°; Max 313°	-30 to +40° C
Survival T	K	Min 238°; Max 318°	-35 to +55° C
Cleanliness			
EMC requirements	N/A	Normal S/C, not sensitive	
DC magnetic	N/A		
Particulate	N/A	Class 10,000 clean room at all times	
water, hydrocarbon, etc	N/A	MCPs very sensitive	Purge through testing and in fairing highly desirable
Miscellaneous			
Mechanisms	No.	Open once door covers	Pyro or bimetal motors
Alignment		0.5° total stability, knowledge	
Orbit requirements		SO orbit is ideal	
AIT/AIV requirements		N/A	

Development approach / schedule

Preferred model philosophy	e.g. PT, EM/QM, FM + FS
Estimated development time	PT (1 yr), EM/QM (2 yr), FM (1 yr), FS (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	**Remarks, proposed risk-mitigating measures.**
Electro-optics	Electrostatic sweeping optics, with sensitivity adjustments need development.
SSD electronics	Highly preferable to develop integrated readout electronics for SSD pixels.
AMP/TOF chip	Highly desirable to meet mass and power requirements

3.2 Radio and Plasma Wave Analyzer (RPW)

3.2.1 Instrument Description

SCIENTIFIC DRIVERS

In-situ wave investigations on many spacecraft (e.g., Ulysses, Wind) have amply demonstrated the crucial role of wave observations in the study of a broad range of solar terrestrial phenomena. Wave observations have been particularly useful in correlation with SOHO optical and EUV observations. Typical waves instrumentation provides measurements of both the electric field and magnetic field in a broad frequency band, usually from a fraction of a Hertz up to several tens of MHz, covering characteristic frequencies in the solar corona and interplanetary medium. Both electrostatic waves and electromagnetic waves can be measured, leading to different diagnostics: electrostatic waves mostly provide *in-situ* information in the vicinity of the spacecraft, while electromagnetic waves can provide an extensive remote sensing of energetic phenomena in the solar corona and interplanetary medium.

Solar radio astronomy in a broad range of frequencies provides a unique means to monitor, track and analyse energetic phenomena taking place in the solar corona and interplanetary medium, in association with solar activity. Radio emission is essentially generated through the interaction of energetic electrons (produced in solar flares and shock waves) with the ambient solar wind plasma. Most radiation mechanisms produce radiation at a frequency close to the local plasma frequency which is directly proportional to the square root of the electron density N_e , leading to the strong frequency-distance dependence shown in Table 3.3.

	electron density Ne (cm^{-3})	**radio frequency**
Low corona	~ 10^8 – 10^{10}	100 MHz – 1 GHz
Medium corona	~ 10^6 – 10^8	10 MHz – 100 MHz
~ 10 Rs	~ 10^4	~1 MHz
~ 40 Rs	~ 10^3	~ 300 kHz
~ 1 AU	~ 10	~ 30 kHz

Table 3.3: Radio frequency as function of electron density.

Using proven techniques, the Solar Orbiter mission profile will allow a radio and plasma wave package to face quite different and new situations: access to waves and turbulence that occur much closer to the Sun; access to viewing angles from well out of the ecliptic plane, permitting for the first time studies of the north-south symmetry of the radio radiation in the solar corona.

The Radio and Plasma Waves Analyser (RPW) comprises two main sub-systems: the Plasma Waves Analyser (PWA) covering *in-situ* measurements and the Radio Spectrometer (RAS) for remote sensing. The two sub-systems share some of the sensors (essentially the low frequency boom antenna) and have a common digital signal processing unit and interface with the spacecraft.

The same receivers can be used to analyse the different types of waves detected by different sensors, for instance electric antennae and magnetic coils. There is a lot of experience in the design of the sensors and receivers, as well as in the analysis techniques. Also all aspects of electromagnetic cleanliness, are well understood and are no longer a problem for modern spacecraft designs (e.g. Ulysses, Wind/Polar, Cassini, IMAGE, Cluster).

INSTRUMENT CONCEPT

Concept of the PWA

The PWA will identify the various plasma waves and kinetic modes comprising the high-frequency part of the fluctuation and turbulence spectra. We provide below a brief description of the characteristics of the plasma waves expected to be present at 0.2 AU.

The expected field strength may range between a few μV/m and mV/m, and up to about 1 V/m for the convection electric field. The magnetic field strength is expected to vary between a few nT and μT, with large differences between the longitudinal and transverse components with respect to the mean magnetic field and the solar wind flow direction (substantial Doppler shifts are to be expected).

From Helios observations one can estimate that a sensitivity equal to 10^{-6}nT/(Hz)$^{1/2}$ will be required at 0.2 AU to identify without ambiguity whether the observed waves are electromagnetic or electrostatic.

At and above the electron plasma frequency intense electron plasma oscillations and solar radio waves are expected. Helios observations show that wave intensities associated with type III bursts increase very rapidly with decreasing radial distance from the Sun. Intense emission could extend up to 10^{-4} nT/Hz$^{1/2}$.

The measurement requirements necessary for the identification and analysis of the various electromagnetic phenomena expected to be measured at 0.2 AU are given in Table 3.4.

Phenomenon	**Frequency band**	**Minimum amplitude (nT/Hz$^{1/2}$)**	**Maximum amplitude (nT/Hz$^{1/2}$)**
Doppler-shifted ELF/VLF waves and turbulence	1 Hz – 10 kHz	10^{-5} at 10 Hz	10^{-1} at 10 Hz
Ion acoustic waves and Radio waves	10 kHz – 10 MHz	A few 10^{-7} at 1 MHz	10^{-4} at 1MHz

Table 3.4: Measurement requirements for the fluctuating magnetic field.

The PWA will cover a broad band in frequencies, extending from about 1 Hz into the MHz range. Resolving the vector components of the electric and magnetic fields is scientifically highly desirable to determine wave modes unambiguously. However, due to different constraints of the mission, we propose to measure one electric field component only, with a boom antenna mounted in the anti-Sun direction, i.e. in the shadow of the spacecraft. The three components of the fluctuating magnetic field can be easily measured with a 3-axial search coil magnetometer arranged in a compact configuration and mounted on a short boom which should also point into the anti-Sun direction. This boom is required for magnetic cleanliness reasons and could be shared with the MAG (see 3.2.4) as long as a minimum distance between the two sensors is respected. There is a strong heritage for those sensors mounted on three-axis stabilised spacecraft, e.g. on Galileo, Cassini, STEREO. The PWA will also benefit from studies and developments under progress for the Solar Probe.

The PWA will perform on-board processing of the data and deliver selectable wave forms (Time Domain Sampler, TDS), to identify non-linear coherent fluctuations and correlate them with measurements from the EPD.

Concept of the RAS

The RAS will measure the solar and interplanetary radio waves in the frequency range from 100 kHz to 1 GHz, with a sweep period between 0.1 s and 10 s and a high spectral resolution ($\Delta f/f \approx 0.07$). The RAS will observe plasma processes associated with energetic electrons from the low corona up to about 0.5 AU. It will probe the plasma at distances ranging from the solar surface to the spacecraft location, thereby connecting the low-altitude coronal regions observed by the optical instruments with the near-Sun heliospheric conditions specified by the in-situ measurements. Since radio radiation is generally beamed (beam widths sometimes down to a few tens of degrees) more or less along a radial direction from the Sun, this technique is particularly relevant for different vantage points, for instance when the Solar Orbiter observes the far side of the Sun. The time history provided by the regular acquisition of the radio dynamic spectrum will help to trace the development of an active region in a synthetic manner. Activity indices could thus be produced.

The time resolution required to detect the rapidly varying solar bursts varies with the radio frequency. Typically, the duration of a type III burst (stream of energetic electrons) is $\Delta t(s) = 220 / f(\mathrm{MHz})$. This points to time resolutions of the order of 0.1 s or better for the high frequencies and of 10 s for the low frequencies.

A single boom antenna can be used for the low frequency part of the band (below, e.g. 20 MHz), where the sensitivity is essentially determined by the sky background radiation. Above 20 MHz and particularly above 200 MHz, the background is determined by the receiver temperature and gain is needed in order to detect solar radio bursts: a simple boom cannot be used anymore. A broad-band antenna with significant gain is thus required. Each one of the receivers will be fed by different antenna or search coil systems.

The RAS instrument could be divided into three sub-spectrometers, for instance:

- RAD1: 100 kHz - 16 MHz
- RAD2: 16 MHz - 200 MHz
- RAD3: 200 MHz - 1 GHz

Different designs are used on many space missions and the heritage goes right back to the birth of space radio astronomy at long wavelengths.

RPW instrument description

An overall block diagram of the Solar Orbiter RPW instrument is shown on Figure 3.2.

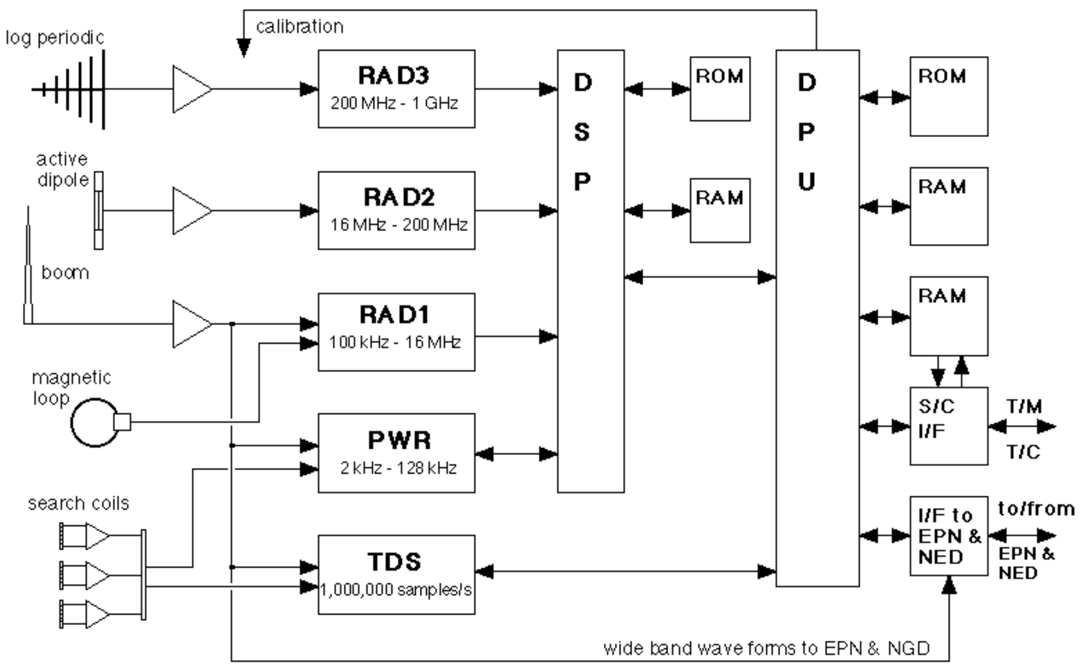

Figure 3.2: Block diagram of the Solar Orbiter RPW instrument.

Plasma waves and low frequency activity will be monitored by a receiver covering the band from about 1 kHz up to 100 kHz. In order to allow for a high time resolution, whilst minimising resources, it can be of the type flown on Wind and planned for STEREO, using a digital filtering technique.

The wave form analyser or Time Domain Sampler (TDS) is a broad band waveform sampler with 4 channels (1 electric field component and 3 magnetic field components). The signal from the boom antenna and search coil are sampled at a rate up to 10^6 samples per second. Events are detected on board and transmitted according to the available bit-rate. This is done using "intelligent" algorithms which can return a quality value for the events. A number of random events can be sent to the ground without regard of their quality in order to provide unbiased statistics. Real time triggers can be sent to the EPD for special studies of wave-particle phenomena.

The radio receivers can be of the classical super-heterodyne type. Frequency synthesisers will allow for a maximum flexibility in the choice of the observing frequencies (for instance to avoid "polluted" frequencies on the spacecraft). Each sub-receiver can consist of up to 256 selectable channels. Only a selection of these channels could be transmitted to the telemetry stream.

The DPU will command the different subsystems and interface with the spacecraft. The variety of instruments call for a packetised telemetry system, that can be managed by the DPU itself, as is done for similar packages on the Wind spacecraft (fixed format telemetry system) and STEREO. The DPU will interface with other instruments on board the spacecraft, particularly the EPD. The DPU could be shared with some other instruments from the payload, e.g. with MAG.

Tables 3.5a, 3.5b, 3.5c provide the main characteristics of the sensors and analysers comprised in the RPW investigation. Numbers are indicative and can vary according to design options

	Magnetic sensors	**frequency band**	**Sensitivity ($nT/Hz^{1/2}$)**	**Size**	**mass (g)**	**power**
B1	Search coils	1 Hz – 10 kHz	10^{-5} at 10 Hz	3 rods, 18 cm each	500	0.2c500
B2	Magnetic loop (optional)	10 kHz – 10 MHz	10^{-6} at 1MHz	20 cm diameter	600	0.2
Total					*1100*	

Table 3.5a: Main characteristics of the RPW magnetic sensors.

	Electric sensors	**frequency band**	**Size**	**mass (g)**	**power**
E1	Radio boom	1 Hz – 20 MHz	5 m boom	1300	N/A
E2	Active dipole	20 MHz – 200 MHz	~ 1m x 2cm x 2 cm	800 (TBC)	0.3
E3	Log-periodic antenna	200 MHz – 1 GHz	~ 1m x 1m x 1cm	1500 (TBC)	N/A
Total				*3600*	

Table 3.5b: Main characteristics of the RPW electric sensors.

Analyser	**frequency band**	**Sensor(s)**	**Sensitivity**	**mass (g)**	**Bitrate (b/s)**	**Power (W)**
TDS	wave form 1 MHz	E1, B1	30mV/m, 0.2 pT	800	1700	1.0

PWR	2 kHz – 128 kHz	E1, B1	10 nV/Hz$^{1/2}$, 10^{-5} nT/Hz	800	1250	0.7
RAD1	100 kHz – 16 MHz	E1, (B2)	10 nV/Hz$^{1/2}$, 10^{-6} nT/Hz$^{1/2}$	800	800	0.8
RAD2	16 MHz –200 MHz	E2	~ 10^{-20} W m^2 Hz^{-1}	700	600	0.7
RAD3	200 MHz – 1 GHz	E3	~ 10^{-20} W m^2 Hz^{-1}	700	600	0.7
DSP/DPU				1200		1.5
Converter				800		1.2
Harness etc.				800	50 (H/K)	
Total				*6600*	*5000*	*6.6*

Table 3.5c: Main characteristics of the RPW analysers and subsystems

INSTRUMENT SUMMARY

- triaxial search coil mounted on the magnetometer boom,
- magnetic loop antenna,
- boom antenna (5 to 6 m on side opposed to Sun),
- VHF/UHF system comprising an active dipole and a log-periodic antenna.

Role of main subsystems:

- RPW/PWA: identify plasma waves and kinetic modes (high frequency part of the fluctuation and turbulence spectra)
- RPW/RAS: measure and track solar and interplanetary radio bursts; connect low-altitude coronal regions observed by the optical instruments with the *in-situ*, local solar wind conditions at the spacecraft, study directivity of radio emissions

Instrument	Mass (kg)	Power (W)	Volume (cm^3)	Data rate (b/s)
Total RPW	11.3	7.3	TBD	5000

Table 3.6: RPW resource requirements.

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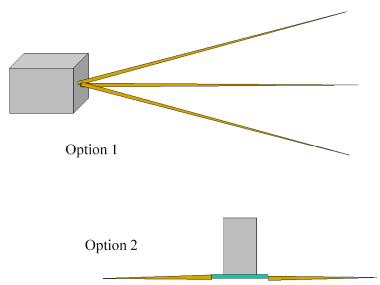
3.2.2 Orbit, Operations and Pointing Requirements

The antenna systems would have to be pointed to the Sun (accuracy of a few degrees) and thus protected from the direct sunlight by a shield not affecting the radio wave propagation.

3.2.3 Accommodation



These antenna systems can be grouped together and mounted on the same boom in order to have a common mechanical (deployment mechanism) and electrical interface with the spacecraft. Interface and Physical Resource Requirements

Antenna systems

Antenna systems for the range from 20 MHz to 1 GHz are readily available off the shelf for ground based systems, including "portable" ones. Different types and techniques are available, and the theory of the measurement is well known. They would have to be developed specifically for the Solar Orbiter mission. The RAD2 band (16 MHz - 200 MHz) could be covered by an active dipole antenna and the RAD3 band (200 MHz - 1 GHz) by a log-periodic antenna. The whole system can be kept to a reasonable size (longest element of the order of 1 m) and provide the required gain. The beam shape of the antennae mounted on the spacecraft will have to be modelled and measured using existing techniques.

Where can we accommodate three 5-6 meters stacers/booms, with the preferable option of their bases (contacts to S/C) connected to the same box. This is option 1 on the figure. Note that the angle between antenna and their lengths are TBD. We can study that from the scientific requests on these parameters.

3.2.4 Cleanliness, Ground Operations and Other Requirements

Electro-Magnetic Cleanliness (EMC)

Many of the RPW scientific objectives are based on the measurement of low-level signals for which maximum sensitivity is required. Several inexpensive measures can be taken at the spacecraft and project level to ensure that the Solar Orbiter spacecraft is clean from the point of view of both conducted and radiated electromagnetic interference. The sensitivity of the RPW instrument will be approximately 10^{-8} $V/m/Hz^{1/2}$ in the frequency range of 1 kHz to 20 MHz and 3×10^{-7} $V/m/Hz^{1/2}$ below 100 Hz. In addition, the wave form analyser is sensitive to impulsive interference of duration as short as a fraction of a microsecond. Although these sensitivities may appear to require an excessively "clean" spacecraft, they are not difficult to achieve if good EMC practices are incorporated in the spacecraft design.

3.2.5 Open Points and Critical Issues

Some sharing of the processing capabilitiy with STYX could be envisaged.

Name / acronym	**Solar Orbiter Radio and Plasma Wave (RPW) Instrument**

Objectives	**To measure electrostatic and electromagnetic waves in the range ~DC to a few tens of MHz** - Plasma waves & kinetic modes : HF part of fluctuations, turbulence, particle/wave interactions - Electromagnetic (radio) waves : remote sensing of electron energetic phenomena in corona & IP medium, measure and track of solar & interplanetary radio bursts

General description	**RPW is composed electric antennas and magnetic sensors connected to pre-amplifiers and analysis electronics consisting of LF & HF receivers and analyzers, and waveform samplers**

Reference P/L and/or heritage	**similar packages on Helios, ISEE, Ulysses, Wind, Cassini, STEREO**

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Electric boom antenna	$V/m/Hz^{1/2}$	1 to 3 booms, 5 - 6 m	3 booms will allow both common mode rejection and 3-vector electric field, radio polarization, and direction-finding measurements
Active dipole	$V/m/Hz^{1/2}$	~ 1m × 2 cm × 2 cm	accommodation concerns
Log. Periodic antenna	$V/m/Hz^{1/2}$	~ 1m × 1 m × 1 cm	accommodation concerns
Search coils	$T/Hz^{1/2}$	3 rods, 18 cm each	
Magnetic loop	$T/Hz^{1/2}$	1 loop, 20 cm diameter	frequency range TBD
Physical			
Mass, total	kg	10.2 to 11.3 kg	
Mass of electric sensors (booms+active dipole+Yagi)	kg	3.6 kg (one elec. boom) 4.5 kg (3 elec. booms)	Using STEREO boom mass (1.8 kg/element) and TBC A.D. and Yagi masses
Mass of magnetic sensors	kg	1.1 kg	
Mass of receivers and analyzers - TDS (time domain sampler) - PWR (plasma wave receiver) - RAD1 (radio receiver 1) - RAD2 (radio receiver 2) - RAD3 (radio receiver 3) - DSP/DPU - Converter - Harness	kg	 0.8 kg 0.8 kg 0.8 kg 0.7 kg 0.7 kg 1.2 kg 0.8 kg 0.5 kg	 Waveform samples 2 – 128 kHz 100 kHz – 16 MHz 16 – 200 MHz 200 – 1000 MHz DPU could be shared with other exp.
Power			
Average	W	7.5 W	
Peak power	W		
Stand-by	W		
Data rate / volume			
Average data rate	Bits/sec	5000 bits/sec	
Peak data rate	bits/sec		
Data volume /orbit	Mbits	47520	5 kb/s x 3600 x 24 x 110 days / orbit ?
Own data storage	Mbits		
Thermal			
Other requirements	N/A	**Study heat conducted**	

		and radiated by electric antenna booms back to S/C	
Cleanliness			
EMC requirements	N/A	**Everywhere for S/C & instrument payload**	
DC magnetic	N/A		
Particulate	N/A		
Miscellaneous			
Mechanisms	No.	No. and type, shutter, filters, etc.	

Development approach / schedule

Preferred model philosophy	e.g. EBB, EM, QM, FM + FS
Estimated development time	EBB (1 yr), EM (1 yr), QM (2 yr), FM (1 yr), FS (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
Accommodation	Accommodation of electric antenna booms, magnetic sensors (1 boom for all mag. sensors, boom shared with MAG)
Thermal concerns	thermal concerns associated with extending conducting antennas into sunlight, including antenna material properties
EMC program	A good EMC program will be necessary for a successful implementation of RPW

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
Receivers			Need miniaturization effort (ASICs)
Antennas			Antenna materials must be studied for high temperatures

<u>Technology Readiness Level (TRL)</u>:
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

<u>Design Maturity Level (DML)</u>:
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

3.3 Coronal Radio Sounding (CRS)

3.3.1 Instrument Description

A passive radio science experiment can be carried out using the available radio links. The spacecraft will pass behind the Sun (superior solar conjunction) on many occasions, making it possible to investigate the solar corona by radio sounding at solar distances much less than the perihelion distance and down to at least 2 solar radii. Integrated line-of-sight parameters such as electron content (densities), Faraday rotation, scintillations and angular broadening can be recorded. The Solar Orbiter orbit and the proximity of the spacecraft to the Sun will create unique situations for this kind of analysis. The CRS investigation with a two-way radio link via the spacecraft high-gain antenna (ranging capability) would require a radio subsystem with dual-frequency phase coherent downlinks at X-band and Ka-band. Linear polarisation of the downlink signals would enable Faraday rotation measurements as an option. The RF power for both radio links (X and Ka) planned for this mission is sufficient for this investigation. A two-way dual-frequency coherent radio link (X-band uplink; X-band and Ka-band simultaneous coherent downlinks) is considered as the optimal configuration for a sufficiently stable link and for detecting signatures of CME events traversing the radio ray path.

It is presently unclear whether or not an ultra-stable oscillator would be necessary to guarantee the frequency stability of the downlinks. If only a one-way radio link (spacecraft to ground) is feasible due to operational constraints during solar conjunction, the transmitted radio signal must be stabilised by an onboard ultra-stable oscillator (frequency stability 10^{-11} to 10^{-12} at 3 s integration time, 200g, 3 W). The practicality of using linear polarised signals for Faraday rotation measurements is another trade-off to be explored. The final decision depends critically on the geometry and mission operation plan. Considerable interaction between the radio scientists and the spacecraft’s radio subsystem contractor is a prerequisite for conducting a successful coronal sounding experiment. The occultation geometry during the solar conjunctions is another aspect that needs to be studied in detail.

3.3.2 Orbit, Operations and Pointing Requirements

None

3.3.3 Accommodation

N/A

3.3.4 Interface and Physical Resource Requirements

N/A

3.3.5 Cleanliness, Ground Operations and Other Requirements

N/A

3.3.6 Open Points and Critical Issues

The design of the high-gain antenna by the S/C contractor should taken into account the following requirements:

- radio subsystem with dual-frequency phase coherent downlinks at X-band and Ka-band.
- linear polarisation of the downlink signals as an option.

3.4 Magnetometer (MAG)

3.4.1 Instrument Description

SCIENTIFIC DRIVERS

Measurement of the solar wind magnetic field is essential to the Solar Orbiter in situ science package, as well as several of the mission goals to investigate links between the solar corona and the heliosphere. The magnetic field is a key property of the solar wind on both magnetohydrodynamic (MHD) and kinetic scales, and the magnetometer investigation on Solar Orbiter must measure it with high precision (a few pT) at sub-second resolution – see the table below for details of the requirements of different scientific goals.

The link between coronal structures and their signatures in the solar wind is currently poorly understood. By travelling to 0.21 AU, and imaging coronal structures at the same time as measuring the properties of the solar wind emanating from them, Solar Orbiter will help to join up our knowledge of coronal and solar wind structures and physics. In addition, near co-rotation with the corona will allow the separation of the effects of temporal and spatial variations in solar sources. Of particular interest are coronal mass ejections, coronal hole flows (particularly their edges), sector boundaries and polar plumes. The magnetic field signatures of all these structures must be measured by the Solar Orbiter magnetic field investigation. The precision (a few degrees in angle, around 1nT absolute) and time resolution (a few seconds) required are not onerous for contemporary magnetometers, however. Comparison of magnetic fields at different points in the spacecraft orbit (for example, different latitudes or solar distances), and with those measured by other spacecraft throughout the heliosphere, requires stability of measurements, and in particular the accurate determination of spacecraft and instrument contributions to the measured fields.

Measurements of the solar wind plasma by onboard instruments – vital for both MHD and kinetic scale physics – requires knowledge of the local magnetic field orientation. The magnetometer must provide this vector, with a precision of a few degrees, to enable the plasma instruments to make reliable moment and distribution measurements.

The properties of waves and turbulence in the solar wind and their relation to the coronal sources of solar wind will provide important information on coronal heating processes, MHD turbulence and energetic particle propagation through complex magnetic fields. Magnetic field variations must be measured routinely on all MHD scales (up to spacecraft timescales of less than a second) with sufficient precision (a few pT) to quantify their variations on these scales.

Solar Orbiter will be able to study kinetic effects in the solar wind plasma, which will build on earlier Helios measurements and at the small solar distances near perihelion will provide important information about solar wind acceleration processes, by quantifying the full particle distributions and how these interact with the magnetic field. To achieve these goals, the magnetic field must be measured at a time resolution significantly higher than the proton gyro-period (typically tens of Hertz at perihelion), necessitating up to 128 sampled vectors/s– a rate which is only achievable using a burst mode. Indeed, investigation of kinetic physics is the primary driver for time resolution and precision for the Solar Orbiter magnetic field investigation. Telemetry restrictions mean that strategies such as recording long intervals of burst mode data onboard, but only downloading shorter intervals selected on the ground, may prove necessary.

Mission goal	Timescale required	Precision required
Kinetic physics	128 vectors/s (burst mode)	1nT absolute, few degrees angular, few pT resolution
Large scale solar wind structure	1 vector/s (normal mode)	1nT absolute, few degrees angular
MHD waves and turbulence	16 vectors/s (normal mode)	Few pT resolution
Onboard plasma moments	Plasma instrument resolution (few s)	Angle only: to within a few degrees

Table 3.7: MAG scientific requirements

INSTRUMENT CONCEPT

The magnetometer may consist of a tri-axial sensor system. A magnetically clean spacecraft will be required. Though this should not be a cost-driver for the spacecraft, is does drive the design to some extent and therefore considerations of magnetic cleanliness should enter into the system design as early as possible.

The heritage of MAG includes dozens of similar instruments flown on various magnetospheric, planetary and deep space missions. The MAG sensor and electronics for the Solar Orbiter require no new technology developments. Nevertheless, there are a number of implementation issues which need to be addressed; these are listed in section 3.4.6.

State of the art instruments can be built for approximately 1 kg, and further mass savings may be possible given time and resources for the development.

MAG might consist of a 3-axis fluxgate magnetometer with the sensor mounted at the end of a deployed boom, which is positioned in the shadow of the Orbiter. It measures fields in several gain-ranges, which are automatically selected by the DPU according to the in-situ magnetic field strength. The DPU of MAG is expected to be shared with the rest of the in-situ payload.

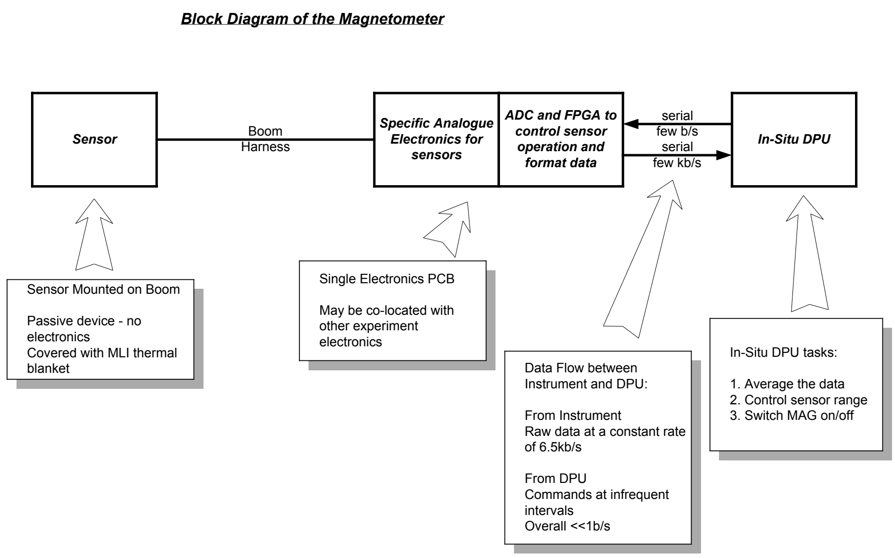


Figure 3.3: Block diagram of the MAG

DESIGN OPTIONS

The main design option which can be considered is to add an additional sensor (see section 3.4.6). The principal advantage is that this provides some possibility to resolve the spacecraft magnetic field interference. This may be necessary in case there is no magnetic cleanliness programme for the spacecraft.

For this option, the mass and power resources required would be increased by approximately 50%.

3.4.2 Orbit, Operations and Pointing Requirements

The MAG has no specific requirements on orbit or pointing.

Operationally, the MAG is a simple instrument. The DPU will control the MAG ON/OFF status and select the sensor gain-range according to a pre-defined 'auto-ranging' algorithm. The MAG has a built-in self calibration function which will be used occasionally (approx once per week), but this will be planned in advance and executed by command. The MAG data rate is switched between Burst, Normal and Survey modes according to the mission-plan or in response to some on-board event trigger.

It is expected that MAG data will be provided to other instruments on-board and/or that some processing of the MAG data in the DPU will be performed in order to select 'events' in the magnetic field data. These may be used to trigger other experiments' burst modes.

3.4.3 Accommodation

ELECTRONICS

The electronics of the MAG sensor may be implemented on a single PCB comprising the sensor analogue electronics, A to D converter and a digital interface to the DPU. This electronics can be housed within the enclosure of either the DPU or another experiment's electronics.

SENSOR

The sensor must be mounted at, or towards the end of, a boom in the permanent shadow of the spacecraft. The boom should be of sufficient length to guarantee the sensor is not susceptible to magnetic interference from the spacecraft or other boom-mounted sensor units. It is also critical for the MAG sensor temperature to be controlled with respect to short-term and long-term variations. It is expected that the sensor will therefore be thermally isolated from the boom structure, and surrounded by MLI. Even though the sensor is permanently in shade, due to the very eccentric orbit it is expected that there will be long-term temperature changes. It may even be necessary to implement a heater in the sensor to keep it warm at perihelion. A detailed study of the MAG sensor thermal environment needs to be done (see section 3.4.6)

3.4.4 Interface and Physical Resource Requirements

The mass of the sensor, electronics and harness is approximately 1Kg. Power consumption is around 1.5W, and the required telemetry (normal mode) is about 800 bits/s.

Instrument	triaxial fluxgate magnetometer
Telemetry	256 b/s
Mass	0.88 kg
Power	1.5 W
Special requirements	mounting on short boom (<1 m), in shadow of spacecraft, basic magnetic cleanliness of spacecraft.

Table 3.8: MAG resource requirements.

3.4.5 Cleanliness, Ground Operations and Other Requirements

Cleanliness
The magnetometer should have no specific requirements on cleanliness and contamination control.

Magnetic Cleanliness
For Solar Orbiter, it is not anticipated to have an extensive magnetic cleanliness approach (e.g. Cluster), but experience has shown that good results can nevertheless be obtained. This is not necessarily an expensive or even difficult task, but it needs the cooperation of all concerned - spacecraft, payload and AIV. It is most important that magnetic issues are considered in the design from day one. This can reduce costly workarounds at a later date. Good design practice is quite straightforward, for example minimising the use of permanent magnets and soft magnetic materials, and careful design of power distribution systems to minimise current loops.
Recommendation:
A magnetic cleanliness plan should be drafted. It is assumed that there will be an 'EMC Panel' which will consider general EMC issues, and magnetic cleanliness should be part of their remit. Note that there is considerable experience of this within ESTEC (K. Mehlem) and also within European institutes (e.g. TU-Braunschweig, TU-Graz).

Ground Operations
The magnetometer should have a gain-range which allows operation and functional test to be performed in Earth magnetic field.

A system-level magnetic test is highly recommended since it allows the magnetometer to operate the flight gain-ranges and to check for compatibility with the spacecraft systems. Such a magnetic test may be performed at a large magnetic test facility such as that at IABG, Ottobrun Germany. This test would also be used to verify the spacecraft overall magnetic compatibility versus the requirements.

It is necessary to use non-magnetic tools on the spacecraft and particularly on the boom and associated units.

3.4.6 Open Points and Critical Issues

Thermal Electric Currents
Due to the extreme thermal environment, there may be a problem with currents induced in the (conducting) spacecraft surfaces and structure due to large temperature gradients. The magnetic fields generated would be a special problem close to the magnetometer sensor. Some experimental work has been done (TU-Braunschweig, GSFC) to quantify this. Thermal electric currents can be minimised by 'banding' conducting surfaces which may be subject to large thermal gradients.
Recommendation:
This can be considered as 'magnetic cleanliness'. Advice is needed from the thermal engineers and experts in surface finishes etc.

Thermal Environment of the Magnetometer Sensor
In principle, if the magnetometer is mounted such that it is permanently in the shadow of the spacecraft then the thermal environment can be quite benign. However, this needs to be quantified.
Recommendation:
Advice needed from the s/c thermal engineers. Ideally, a preliminary thermal model of the design will be done which will give a 'first-cut' approximation of the temperatures of the in-situ sensors.

Accommodation of the sensor
A boom is required for the fluxgate magnetometer sensor, in order to keep the sensor as far away as possible from sources of magnetic interference on the spacecraft. Since other in-situ sensors also wish to be boom

mounted, there may be conflicts if there is only one boom. For example, the search coil magnetometer is a very sensitive instrument, and the drive frequency of the fluxgate sensor will probably fall inside its bandwidth. These two units would not normally wish to be co-located. The boom and the sensor units must be in permanent shadow of the spacecraft.

Recommendation:
A long boom should be provided which can accommodate multiple sensor units without interference. A design similar to the STEREO boom should be investigated. This is a 5m boom unit which comprises 1m cylindrical sections which can accommodate a sensor at each junction.

Interference from Spacecraft Systems
The spacecraft design from the assessment study is well optimised from a thermal point-of-view, but both the solar panels and the high-gain antenna are located close to the magnetometer sensor. These are mobile structures which are likely to cause magnetic interference. For the magnetometer this would be impossible to separate from the real in-situ field.
Recommendation:
Request the industry team to study alternatives to the assessment study baseline design.

Magnetometer Calibration
Fluxgate magnetometer sensor calibrations have a tendency to change with temperature in a way which is not entirely predictable or repeatable. On a spinning spacecraft this is less of a problem since the spin-axis offsets can be easily recovered. Solar Orbiter has two problems: firstly it is three-axis stabilised, secondly over an orbit one can expect large temperature changes even for a shadowed unit.
Recommendation
Try to minimise temperature changes at the magnetometer sensor by careful thermal design of the boom and local MLI.
Can the spacecraft execute a series of roll manoeuvres to allow to recover the magnetometer offsets?

Dual Sensor Magnetometer Design
A magnetometer design which uses two fluxgate sensors mounted on the same boom (about 1m apart) would offer some advantages. Firstly, it can help to resolve the spacecraft magnetic field from the ambient field. Secondly, it provides some element of cross-calibration, which can be useful (see previous point on calibration). It also provides redundancy.
Recommendation
In case the accommodation and magnetic cleanliness problems prove intractable, this should be considered as a baseline for the magnetometer design. Note that the resources are not doubled by adding an extra sensor.

Common DPU
The magnetometer has a low processing requirement and can benefit from use of a common DPU and/or a common payload power converter.
Recommendation:
Study common in-situ DPU and PDU

High Temperature Sensor
In case the thermal model indicates that the magnetometer sensor can not be kept at an operating temperature below about 100ºC then it may be necessary to qualify some new sensor materials. There is some work going on already in this field (DSRI/Danish Tech University, TU-Graz).
Recommendation:
TBD on result of s/c thermal modelling.

Name / acronym	**MAG**

Objectives	Determination of in-situ magnetic field vector

General description	Three axis fluxgate magnetometer A single sensor system is suggested in order to minimise resources (mass, power and telemetry). However, a dual-sensor offers increased reliability and aids calculation of the spacecraft magnetic influence.

Reference P/L and/or heritage	Design heritage from many previous missions e.g. Cluster, Cassini, NEAR…

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type		Three axis fluxgate sensor	Three identical ring-cores arranged in an orthogonal triad
Dynamic range	nT	Range / Resolution ±65536nT / 4nT ±2048nT / 125pT ±512nT / 31pT ±128nT / 8pT ±32nT / 2pT	±65536nT range for ground-test. Automatic range switching controlled by DPU. Quoted resolution is digital resolution. Sensor intrinsic noise level (design goal) should be <10pT/√Hz
Operating T	K	170 to 370	Upper temperature limit within the bounds of existing technology, but new developments could give a higher maximum
Configuration			
Physical Units	No	2	
Layout	N/A	An electronics unit mounted on the spacecraft body, and a sensor unit mounted at the end of a non-magnetic boom. The two units are connected by an experiment-provided harness	A suitable boom would be similar to the STEREO design, which is a 5m long design
Location S/C	N/A	Boom-mounted sensor. Electronics mounted on s/c platform.	Electronics close to boom-root to minimize inter-connect distance. The dimensions given for the electronics are for the PCB only, it is assumed that the electronics can be co-located with other payload electronics in a common enclosure
Physical			
Sensor Mass	kg	0.25	Including mounting hardware
Thermal Blankets Mass	kg	0.08	
Harness Mass	kg	0.15	50 g/m + 30g connectors
Electronics Mass	kg	0.40	Sensor electronics + digital interface to DPU
Total Mass	kg	0.88	Does not include enclosure for electronics: assume common housing with other experiment or DPU
Sensor Dimension	cm	11 x 7 x 5	

Harness Length	cm	600	Assume 5m boom plus 1m routing to electronics
Electronics Dimension	cm	15 x 25 x 3	Electronics card only
Power			
Average	W	1.5	
Peak power	W	1.5	
Stand-by	W	1.5	
Data rate / volume			
Average data rate	Bits/sec	816	Down-linked data including HK 16 vectors/sec
Peak data rate	Bits/sec	3264	Down-linked data including HK 128 vectors/sec
Minimum Data Rate	Bits/sec	56	Down-linked data including HK 1 vector/sec
Data volume /orbit	KByte	TBD	
Own data storage	MByte	0	
Thermal			
Electronics Dissipation	W	1.4	
Sensor Dissipation	W	0.1	
Operating T range electronics	K	240 to 330	
Other requirements	N/A	The sensor and electronics temperature should be kept as constant as possible, since temperature variations change the instrument calibration	The primary driver is to keep the sensor temperature as constant as possible.
Cleanliness			
EMC requirements	N/A	No special requirement	
DC magnetic	N/A	<1nT at Sensor position	System magnetic cleanliness approach
Particulate	N/A	No special requirement	
Miscellaneous			
Mechanisms	No.	None	
Alignment		Knowledge of sensor axes w.r.t. s/c reference axes to better than 0.2 degrees	Boom in deployed configuration
Orbit requirements		None	
AIT/AIV requirements		Use non-magnetic tools for AIV	

Development approach / schedule

Preferred model philosophy	EBB, EM, QM, FM + FS
Estimated development time	EBB (0.5 yr), EM (1 yr), QM (0.5 yr), FM (1 yr), FS (0.5yr).

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
New sensor technology	None
Thermo-electric currents in sensor structure, or boom	Risk: Thermal gradients in sensor housing, thermal insulation or boom can cause magnetic disturbance through thermal-electric currents Mitigation: Analysis required to quantify the problem. Careful design of electrically conducting surfaces to avoid large current loops.

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
Sensor	6	2	Basic design of sensor can be identical to that flown on previous missions, but
Electronics	7	2	Electronics can use existing design and components
Harness	6	2	Suitable cable exists

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

3.5 Energetic Particle Detector (EPD)

3.5.1 Instrument Description

SCIENTIFIC DRIVERS

The Solar Orbiter will allow us to study the sources, acceleration and propagation of solar energetic particles in association with coronal and interplanetary shocks. Energetic pick-up particles originating from outgassing or sputtering of near-Sun dust should be measured as well in conjunction with the plasma measurements. It is essential that all measurements are done fast, typically at 1 s, with complete angular coverage to resolve the pitch-angle distributions. The EPD will

- determine *in-situ* the generation, storage, release and propagation of different species of solar energetic particles in the inner heliosphere;
- identify the links between magnetic activity and acceleration on the Sun of energetic particles, by virtue of combined remote-sensing of their source regions and *in-situ* measurements of their properties, while staying magnetically connected with the acceleration sites during solar co-rotation passes;
- characterise gradual (CMEs) and impulsive (flares) particle events and trace their spatial and temporal evolution near the Sun.

INSTRUMENT CONCEPT

The EPD will determine chemical and charge composition and energy spectra of ions in a wide energy range, from about the typical solar wind energies of a few keV to several 100 MeV/nucleon for protons and heavy ions. Electrons should be measured from 10 keV to 10 MeV. The combination of electrostatic E/Q-analysis with time-of-flight E/M-determination and subsequent direct energy measurement in a solid state detector has been employed in many EPDs in the past and is also a possible design option for the Solar Orbiter.

The specific design suggested here resembles closely the one recently selected for the IMPACT instrument on the STEREO mission. It is a multi-head sensor system using solid state technology. Ongoing research in this field is promising and will certainly lead to further improvements concerning energy thresholds and noise level as well as to size reduction.

3.5.2 Orbit, Operations and Pointing Requirements

3.5.3 Accommodation

The measurement requirements can be met readily by a low-mass and compact sensor assembly with different lines of sight and modest aperture sizes.

3.5.4 Interface and Physical Resource Requirements

The whole EPD package will have a total mass of 8.68 kg and consume an average power of 10.2 W. The required telemetry rate is 2.8 kb/s.

The characteristics of the EPD could, e.g. resemble the 4 sensors of the IMPACT instrument on STEREO, as given in Table 3.9.

Sensor	Energy (MeV/nucleon)	GF ($cm^2 sr$)	FOV (° x °)	Name
SET	0.002-0.1	TBD	TBD	Supra-thermal Electron Detector
EPT	e: 0.02 - 0.40 p: 0.02 - 7.00	4 x 0.30, 4 x 0.24, 4 view cones		Electron Proton Telescope
SIT	0.03 - 2	0.3, 1 view cone	17 x 44	Supra-thermal Ion Telescope (He – Fe)
LET	1.5 - 40	4.5, 10 view cones	130 x 30	Low Energy Telescope (He – Ni)
HET	13 - 100	2.4, 2 view cones	48 x 48	High Energy Telescope (He – Fe)

Table 3.9: EPD instrument characteristics.

INSTRUMENT SUMMARY

Instrument	Mass (kg)	Power (W)	Volume (cm^3)	Data rate (b/s)
STE	0.35		594	
EPT	1.33		707	
SIT	1.23		2600	
LET	0.75		1440	
HET	0.70		1800	
Scan platform	2.59		32400	
DPU/H-L-VPS	1.73		2250	
Total EPD	8.68	10.2	4560	2800

Table 3.10: EPD resource requirements.

3.5.5 Cleanliness, Ground Operations and Other Requirements

3.5.6 Open Points and Critical Issues

Name / acronym	**Energetic Particle Detector / EPD**

Objectives	1) to measure spectra, composition and pitch angle distributions of ions and electrons between 0.002 and 100 MeV/nucl. (or MeV respectively).
	2) to • determine generation, storage, release & propagation of energetic particles in the inner heliosphere • identify links between magnetic activity and solar acceleration of particles by combining source region remote-sensing and in-situ property measurements • characterize gradual and impulsive particle events; trace spatial & temporal evolution near the Sun
	3) to provide energetic particle distributions in the inner heliosphere

General description	The EPD for the Solar Orbiter model payload consists of 5 separate detector systems with specific measurement tasks to cover the required range of particles and energies. Determination of the pitch angle distributions on a 3-axis stabilized s/c requires either multiple sensor heads at different s/c locations looking into different directions relative to the magnetic field direction or at least one scan or rotating platform covering more than 270^0. Each detector systems to be described below consists of a set of solid state charged particle detectors of various types, thin foils, magnets, and microchannel plates. 1. **STE** (Supra-thermal Electron Detector): Electron flux and anisotropy, 2-100 keV 2. **EPT/EPS** (Electron and Proton Telescope): Flux and angular distributions of electrons 20-400 keV, protons and He nuclei 20-7000 keV/nucl. 3. **SIT** (Supra-thermal Ion Telescope): Flux of He^3 ions 0.015-0.25 MeV/nucl. and ions of mass 2-60 with 0.03-2 MeV/nucl. 4. **LET** (Low Energy Telescope): Flux and angular distributions protons 1.5-3.5 MeV, He^3 1.5-1.6 MeV/nucl. and ions of mass 2-28 with 1.6-40 MeV/nucl. 5. **HET** (High Energy Telescope): Flux of electrons 1-8 MeV, protons 13-100 MeV, He^3 15-60 MeV/nucl., and He 13-100 MeV/nucl. EPD Experiment is self-adaptive, continuously measuring and evaluating. Data rate adaptable to link performance.

Reference P/L and/or heritage	1. **STE** (Suprathermal Electron Detector): Modified flight hardware from SAMPEX, HESSI, ACE, UCB 2. **EPT/EPS** (Electron and Proton Telescope): Modified flight hardware from (SOHO COSPIN/LION / Messenger, Universität Kiel, ApL/JHU 3. **SIT** (Supra-thermal Ion Telescope): Modified flight hardware from SAMPEX, HESSI, ACE, UMD 4. **LET** (Low Energy Telescope): Modified flight hardware from WIND, ACE, ULYSSES, by SSD/ESA, CALTECH, GSFC 5. **HET** (High Energy Telescope): Modified flight hardware from WIND, ACE by GSFC, CALTECH 6. **SCAN Platform:** Modified flight hardware from Cassini, Messenger by ApL/JHU, MPAe **7. DPU/H-L-VPS:** Modified flight hardware from STEREO by UCB

Parameter	**Units**	**Value / Description**	**Remarks**
Sensor / detector			
Type	N/A	Solid state detectors, various types	
Spectral range	Nm/eV	See above for different sensors	
Operating T	K	243 - 303	

Optics / antennas			
Type	N/A	N/A	
FOV	Rad		
Bandpass	Nm/eV		
Pointing	N/A		
Configuration			
Physical Units	No	7	
Layout	N/A	Sensors on scan platform, DPU close by	
Location S/C	N/A	Scan platform with all sensors requires unobstructed view >20^0 -290^0 towards W of sun and +- 50^0 in N-S direction. Alternatively multiple sensor heads at different corners of s/c	
Physical			
Mass, total	kg	8,68	
Mass unit 1	kg	0,35	STE
Mass unit 2	kg	1,33	EPT/EPS
Mass unit 3	kg	1,23	SIT
Mass unit 4	kg	0,75	LET
Mass unit 5	kg	0,70	HET
Mass unit 6	kg	2,59	Scan Platform
Mass unit 7	kg	1,73	DPU/H-L-VPS
Dimension 1	cm	9 X 6 X 11	STE
Dimension 2	cm	26.6 x 26.6 x 15	EPT/EPS
Dimension 3	cm	10 x 20 x 13	SIT
Dimension 4	cm	15 X 12 x 8	LET
Dimension 5	cm	15 X 12 x 10	HET
Dimension 6	cm	27 x 40 x 30	Scan Platform (for sensors 1-5)
Dimension 7	cm	15 x 15 x 10	DPU/H-L-VPS
Power			
Average	W	10,2	
Peak power	W	12,2	
Stand-by	W	10,2	
Data rate / volume			
Average data rate	Bits/sec	2800	
Peak data rate	Bits/sec	5600	
Data volume /orbit	KByte		
Own data storage	MByte		
Thermal			
Heat load to radiator	W	10, dissipated to S/C I/F plate	
Operating T range	K	243 - 303	
Other requirements	N/A		
Cleanliness			
EMC requirements	N/A		
DC magnetic	N/A		
Particulate	N/A		

Miscellaneous			
Mechanisms	No.	none	
Alignment		+- 1^0	
Orbit requirements			
AIT/AIV requirements		radioactive sources for testing, continuous dry N_2 purge	

Development approach / schedule

Preferred model philosophy	EM, FM + FS
Estimated development time	EM (1 yr), FM (2 yr), FS (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
Sensor technology	1. Some types of silicon solid state detectors are vulnerable by extreme levels of sunlight, radiation, and temperatures expected during the Solar Orbiter mission. New technologies are under development and need to be investigated for this application by a development and test program for near sun space missions. 2. Compared to the 1970th Si(Li) detectors are today much worse in quality due to the lack of optimum raw material. A larger batch of raw material specified for production of Si(Li) detectors should be ordered to ensure future supply of high quality detectors for space applications.
Analog to digital channel electronics	The analog and digital electronics for conversion of detector signals to digital information needs higher degree of integration and lower power consumption for Solar Orbiter application. Development program required.
Thermal	Demand for large unobstructed view in combination with narrow operational and storage temperature range requires sophisticated thermal design.

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
1. STE	7	4	See reference P/L and heritage above
2. EPT/EPS	7	4	See reference P/L and heritage above
3. SIT	7	4	See reference P/L and heritage above
4. LET	7	4	See reference P/L and heritage above
5. HET	7	4	See reference P/L and heritage above
6. Scan Platform	7	4	See reference P/L and heritage above
7. DPU/H-L-VPS	7	4	See reference P/L and heritage above

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

3.6 Dust Detector (DUD)

3.6.1 Instrument Description

SCIENCE DRIVERS

The DUD will analyse interplanetary dust particles with respect to their composition and orbital characteristics and determine *in-situ* the distribution, composition and dynamics of dust particles in the near-Sun heliosphere in and out of the ecliptic. The spatial distribution of particles with masses ranging between 10^{-16} g and 10^{-6} g will be determined. DUD will

- help outlining the extent of the dust-free zone around the Sun;
- discover the sources of dust, e.g. from Sun-grazing comets;
- unravel the role played by near-Sun dust for pick-up ions;
- deliver data relevant for an understanding of proto-planetary discs.

INSTRUMENT CONCEPT

The science objectives can be met with an instrument of, e.g. Giotto heritage. The dust experiment may consist of a multiple sensor assembly, with each sensor looking in a different direction to resolve incidence angles. A low-mass dust-detecting element could look as sketched in Figure 3.4. The dust particles enter the detector through a thin aluminium foil and evaporate during impact on a gold target. A set of wires in front of the target plate collects the ions produced. The signal is analysed to deduce the mass and possibly speed of the dust particle.

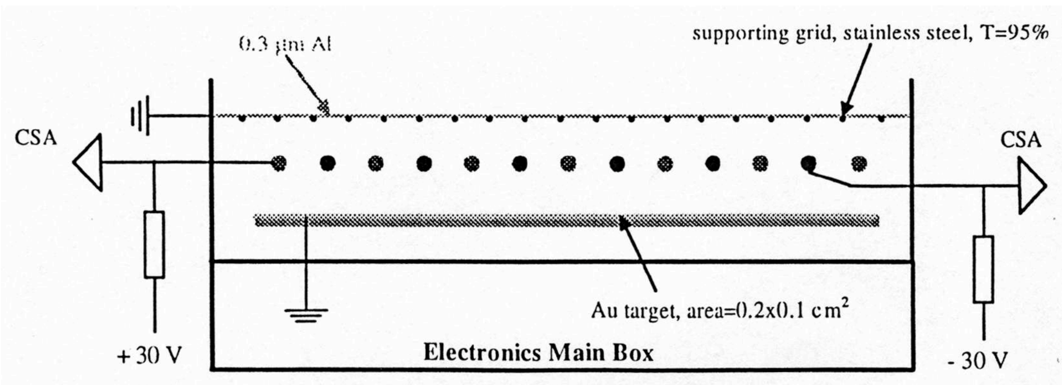


Figure 3.4: Instrument design of DUD.

Full chemical analysis and mass resolution of the important elements H, C, N, O of carbonaceous material and the metals like Na, Mg, Al, Si, Ca and Fe contained in chondritic silicates is desirable but may require a more sophisticated design.

3.6.2 Orbit, Operations and Pointing Requirements

3.6.3 Accommodation

Two identical detector units should be located on the sides of the spacecraft body, one looking 90° off the spacecraft-Sun line in the orbital plane to measure the near-ecliptic dust particles, the other one in the direction perpendicular to the orbital plane in order to measure dust particles in orbits highly inclined to the ecliptic. The field of view of each sensor is ± 80°.

3.6.4 Interface and Physical Resource Requirements

The mass of each sensor is 0.5 kg. Each one consumes a power of 0.3 W and has a volume of 1200 cm^3. The average bit-rate will not exceed 50 b/s. The DPU should be shared with other particle instruments.

INSTRUMENT SUMMARY

Instrument	Mass (kg)	Power (W)	Volume (cm^3)	Data rate (b/s)
DUD	1	0.6	2400	50

Table 3.11: DUD resource requirements.

3.6.5 Cleanliness, Ground Operations and Other Requirements

3.6.6 Open Points and Critical Issues

Name / acronym	Dust detector / DUD

Objectives	
	1) to detect interplanetary dust in the 10^{-15} g to 10^{-6} g mass range
	2) to measure mass distribution and flux rates as function of space and time
	3) to investigate the inner solar system dust cloud and its interaction with the solar wind

General description	Instrument measures electrons and ions that are produced upon the high velocity impact of dust particle.

Reference P/L and/or heritage	Similar measurements on Ulysses, Galileo, Cassini,

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	impact ionization detector	
Spectral range	Nm/eV		
Operating T	K	-30° - 40°	
Optics / antennas			
Type	N/A		
FOV	Rad	□2 □	
Bandpass	Nm/eV		
Pointing	N/A	no pointing, but information about spacecraft attitude with signal required	
Configuration			
Physical Units	No		
Layout	N/A		
Location S/C	N/A	close to spacecraft apex ≠ solar direction	
Physical			
Mass, total	kg	0.400 tbc	does not include DPU
Dimension 1	cm	10 X 10 X 10 tbc	
Power			
Average	W	0.400	
Peak power	W	0.400	
Stand-by	W	tbd	
Data rate / volume			
Average data rate	Bits/sec	4000 tbc	
Peak data rate	Bits/sec	tbd	
Data volume /orbit	KByte	tbd	
Own data storage	MByte	tbd	
Thermal			
Heat load to radiator	W	tbd	
Operating T range	K	tbd	
Other requirements	N/A		
Cleanliness			
EMC requirements	N/A	normal	
DC magnetic	N/A		
Particulate	N/A		

Miscellaneous			
Mechanisms	No.	open cover once	
Alignment		see above	
Orbit requirements			
AIT/AIV requirements			

Development approach / schedule

Preferred model philosophy	e.g. EBB, EM, QM, FM + FS
Estimated development time	EBB (1 yr), EM (1 yr), QM (2 yr), FM (1 yr), FS (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
e.g. new sensor technology	Recent theoretical modeling – To be validated by test – Early BB
e.g. optics with new ARC	Mirror coating technology to be validated via dedicated tests.
e.g. mechanical cooler	Qualification ongoing at specialized company. Qualification by …

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

3.7 Neutral Particle Detector (NPD)

3.7.1 INSTRUMENT DESCRIPTION

SCIENTIFIC DRIVERS

Neutral atoms are closely coupled to the emerging solar wind plasma and give rise to the prominent solar Lyman-alpha corona. The ratio of the densities of neutral hydrogen and protons is small, some parts per million, and the neutral atoms are therefore a trace particle population in the solar wind plasma. Direct observation of the neutral atoms, their flight pass and their density and velocity distribution will help to refine the understanding of the Lyman-α corona, i.e the solar wind acceleration region. The measurement of the density and velocity distribution functions of the hydrogen atoms in the solar wind energy regime, i.e. the hydrogen velocity component parallel to the magnetic field, by an in-situ particle instrument would help to refine Lyman-α corona models. Beyond 3 solar radii, the neutral atoms become more and more decoupled from the plasma. The neutral solar wind constitutes an in-situ trace particle population within the solar wind plasma being observable from Perihelia to Aphelion of the Solar Orbiter orbit.

The neutral atom flux is expected to be about 100-1000 atoms cm^{-2} s^{-1} at 0.21 AU, but it could be up to 10^6 atoms cm^{-2} s^{-1} in a CME. The NPD might also measure energetic neutral atoms emitted from various coronal sources, and thus enable images of these coronal emission regions to be constructed from rays of neutral atoms with different velocities.

INSTRUMENT CONCEPT

Neutral atom detectors have been flown on SOHO, Cassini and IMAGE. Time-of-flight instruments are for example flown on Ulysses, SOHO and ACE.

As a baseline, the instrument parameters of the NPD on the Solar Orbiter could be: Energy/mass range: 0.6 to 10 keV/nucleon, velocity resolution about 0.05, mass resolution is a secondary objective. The field of view of the NPD should be centred at about 7.5° off the spacecraft-Sun-centre line, the field of view half-cone measures 4°.

The instrument entrance system must block radiation emerging from the solar disc. The UV scattered in the solar corona, i.e. Lα, must be suppressed by a factor of more than 10^{-12}. The entrance system must reduce ions and electrons of the solar wind plasma by a factor of about 10^{-9} whilst transmitting neutral atoms.

The neutral atom detector may consist of two sections: the first compartment accommodates the baffle plates (or fast shutters) keeping out solar UV and the deflection unit for keeping off unwanted charged particles. The second section may contain a time-of-flight (TOF) double-coincidence detector. Between the two sections is the pin-hole/slit that defines the optical properties of the neutral atom detector.

The direct solar radiation is blocked by the solar orbiter sunshield (making use of the aberation of particles up to 8°, see Figure 3.5). For the suppression of scattered light within the collimator, the plates could be covered with sawtooth-like structures. The collimator might consist of, alternatively positive and negative, high-voltage biased plates and therefore act as a charged-particle deflector. The collimator plate length, distance and potential would determine the cut-off of the charged-particle transmission (at about 100 keV/q). UV, i.e. Lα, light would be, in part, transmitted through the collimator. The grid in front of a pin-hole should reflect the UV light. The grid could consist of 2 gratings with a grating constant matching the Lα peak. The grids are more transparent for particles than for UV photons (ratio up to 10^5 per grating, but a function of the UV light wavelength, the latter must be suppressed up to about 260 nm). More promising UV rejection should be achievable via fast shutter arrays (e.g. based on bimorph piezos, operating at about 10 MHz).

A possible design of the neutral atom 2-coincidence detector could be: The neutral atoms pass through the pin-hole (or slit) and a 0.5 to 1.1 [?]g/cm^2 carbon foil. Passing through the carbon foil, they release secondary electrons which are accelerated towards a START MCP (multi-channel plate). The atoms travel through a 3-cm-wide TOF drift region and hit the rear STOP MCP. The time of flight for 1 keV/n-100 keV/n atoms is about 8 to 80 ns. The position resolution is 1-D for the START MCP and 2-D for the STOP MCP. The flight pass of the neutral atom can be calculated from these measurements by an inversion of the instrument function. The detector determines the velocity (energy/nucleon) and the flight pass (imaging) of the neutral atoms.

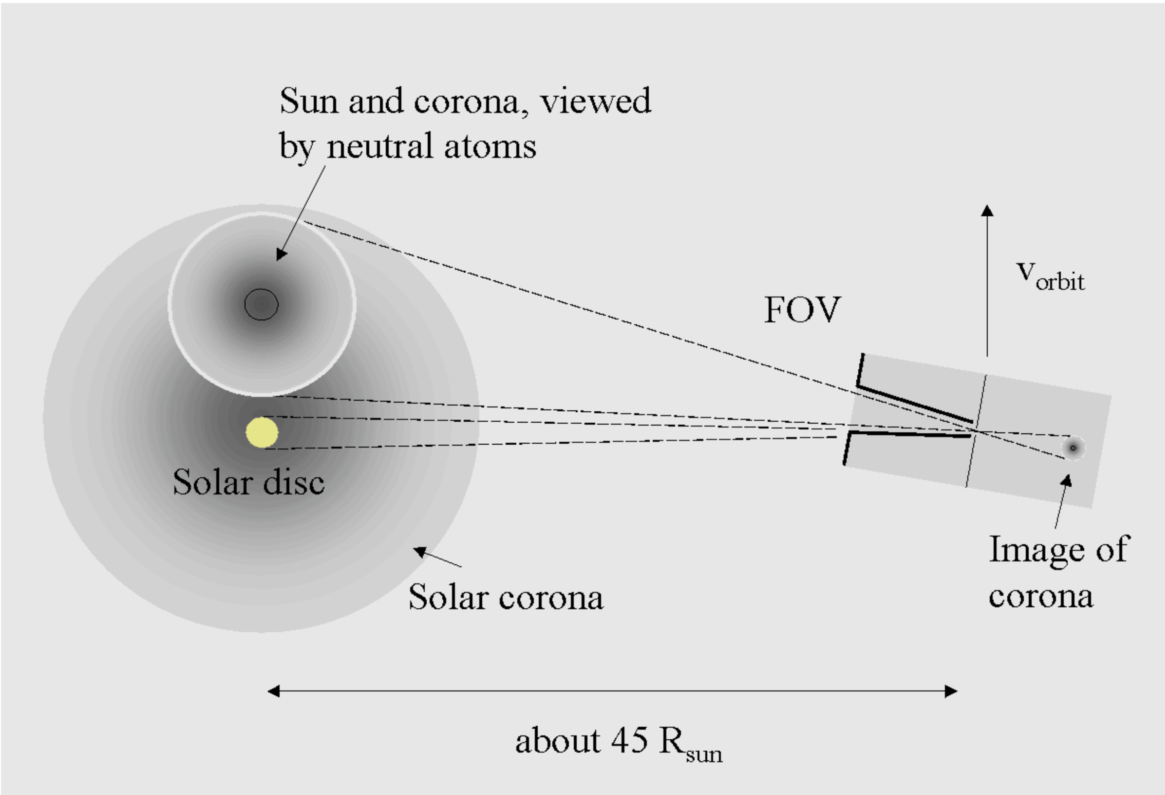


Figure 3.5: Instrument schematics and field of view of the Neutral Particle Detector. Imaging the solar corona via neutral atoms. The aberration causes the shift of the “neutral solar wind” image of the corona off the Ly-[?] corona (plotted for v_{sw} = 400 km/s, v_{orbit} = 65 km/s).

3.7.2 Orbit, Operations and Pointing Requirements

The instrument points 7.5^o off the S/C-Sun axis in the plane of the orbit. The field of view is an half cone of 3.5^o.

3.7.3 Accommodation

The instrument should ponit 7.5° off the S/C - Sun axis in the plane of the orbit. A possible accomodation would be the +Y panel, in a postion behind the sunshield and with an unobscured field of view of +/- 4° , 7.5° off the S/C - Sun axis.

3.7.4 Interface and Physical Resource Requirements

The NPD can be built for a mass of 1.3 kg, including 0.2 kg for a shared DPU. The power consumption would be about 2 W. The required telemetry rate is less than 0.32 kb/s. It is envisaged to use a common DPU and it is possible to share mechanical structure elements with e.g. SWA. The FOV has to be 7.5° off the spacecraft-Sun line because of the typical 5° to 10° aberration at 0.21 AU for the fast and slow solar wind velocities.

INSTRUMENT SUMMARY

Unit	Mass (kg)	Power (W)	Volume (cm^3)	Data rate (b/s)
Structure	0.330			
Electronics	0.430	2		
Harness	0.040			
Connectors	0.020			
Sensor (MCP+collimator)	0.280			
DPU(shared)	0.200			
Total	1.3	2	2560	320 b/s

Table 3.12: NPD resource requirements.

3.7.5 Cleanliness, Ground Operations and Other Requirements

At S/C level a class-100000 clean room shall be enough. The instrument is envisaged to operate autonomously and commanding is only required to update lookup table for voltage settings and data classification schemes.

3.7.6 Open Points and Critical Issues

The possible UV-suppression system via bimorph piezo shutter system is being studied.

Increasing the detection efficiency is being studied.

Name / acronym	**NPD - Neutral Particle Detector**

Objectives	In-situ measurement of neutral solar wind atoms and determination of their velocity and temperature

General description	Neutral particle detector: Goals, objectives Neutral atoms are closely coupled to the emerging solar wind plasma and give rise to the prominent solar Lyman-alpha corona. The ratio of the densities of neutral hydrogen and protons is small, some parts per million, and the neutral atoms are therefore a trace particle population in the solar wind plasma. Direct observation of the neutral atoms, their flight pass and their density and velocity distribution will help to refine the understanding of the Lyman-alpha corona, i.e the solar wind acceleration region. Beyond 3 solar radii, the neutral atoms become more and more decoupled from the plasma. The neutral solar wind constitutes an in-situ trace particle population within the solar wind plasma being observable from Perihelia to Aphelion of the Solar Orbiter orbit.

Reference P/L and/or heritage	Modification / improvement of flight H/W: LENA, MENA on IMAGE, GAS on ULYSSES

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	In-situ particle detector	
mass range energy/mass	amu keV/nuc	hydrogen, helium,oxygen (mass resolution is 2nd objective) 0.6 to 10 keV/n (0.1 to 100 keV/n is 2nd objective) resolution: better than 0.1 (0.05 is 2nd objective)	
Operating T	K	240 to 320 K	
Entrance system			
Type	N/A	Collimator / Sunshield	
FOV	degree	8° full cone	
Pointing	degree	7.5° off Sun-Satellite axis(in X-Y plane) (TBC)	
Pointing error	degree	0.1°	
Configuration			
Physical Units	No	1	
Location S/C		facing sun (X)	
Physical			
Mass, total	kg	1.3	
Mass unit 1	kg	1.1	
Mass unit 2	kg	0.2	shared mass allocation for DPU
Dimension 1	cm	L x W x H 40 x 10 x 10	
Dimension 2	cm	L x W x H N/A	shared mass in central DPU
Power			
Average	W	2	
Peak power	W	3	

Stand-by	W	0.5	
Data rate / volume			
Average data rate	Bits/sec	<320	
Peak data rate	Bits/sec	N/A	
Data volume /orbit	KByte	estimate: $3*10^5$	
Own data storage	MByte	less than 0.1	
Thermal			
Heat load to radiator	W	estimate: smaller than 8 W (TBC)	
Operating T range	K	240 to 320	
Other requirements	N/A	N/A	
Cleanliness			
EMC requirements	N/A	Surface of detector (TBC)	
DC magnetic	N/A	not required	
Particulate	N/A	Clean room class 10000	
Miscellaneous			
Mechanisms	No.	TBD.	
Orbit requirements		N/A	
AIT/AIV requirements		N/A	

Development approach / schedule

Preferred model philosophy	e.g. EBB, EM, QM, FM + FS - yes
Estimated development time	EBB (1 yr), EM (1 yr), QM (2 yr), FM (1 yr), FS (1 yr). yes

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
new sensor technology	detector efficiency requires new technology
	.i.e surfaces
	suppression of scattered UV photons (example: Ly-alpha photons) and solar wind ions by about 10^12 required

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
Detector	2/1*	4	Based on heritage / *New concepts are still looked into
Electronics	2	3	as plasma instruments
Entrance System	2	3/4	UV suppression / Ion filter

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

3.8 Neutron Detector (NED)

3.8.1 INSTRUMENT DESCRIPTION

SCIENTIFIC DRIVERS

Solar neutrons (and γ rays) are a by-product of solar flares due to spallation reactions between energetic protons and α-particles. The overwhelming majority of these neutrons are thermalised within minutes and later absorbed by hydrogen to form deuterium and the well known 2.2 MeV γ line. Some neutrons, however, escape into space and can eventually be detected (before they decay with a half-life of 619.9 s) by a sensor located in close proximity to the Sun. The temporal correlation of neutron fluxes with solar energetic particle fluxes and with optical observations in the EUV provides valuable information on the solar particle acceleration process, which is inaccessible by other means.

The advantage of neutrons over charged particles not being delayed by travel along complicated paths through magnetic fields will allow an unambiguous temporal correlation with optical observations of features of flares in different energy ranges.

Including the NED on the Solar Orbiter offers routine monitoring of neutron fluxes and, optionally, γ-rays and will be a "first" in solar science. A major contribution to the solar γ-ray flux stems from the 2.2 MeV line, which is due to the reaction of thermal neutrons with protons to form deuterium. Hence, the γ-ray detector registering "extinct" solar neutrons would ideally complement the neutron detector, which is to record "live" neutrons in-situ.

INSTRUMENT CONCEPT

Following the design of the instrument proposed for the Solar Probe the heart of the suggested NED instrument consists of a cell containing thin (<10 μm) thorium foils. With a cross section of typically 100 mbarn (at energies above 1 MeV) an efficiency of 10^{-5} for producing fission fragments - to be detected in a pulse ionisation chamber - can be achieved. The anti-coincidence chamber surrounding the sensitive volume contains a counting gas, e.g. methane or argon, and energetic particles which might produce false counts in the ionisation chamber will be detected with an efficiency of close to 100%. The angular resolution of the NED will not be sufficient to map individual features on the solar surface.

Optionally, incorporation of a facility to monitor solar γ-rays into the experiment would greatly enhance the scientific return.

Planetary flybys could be used to block the solar neutron flux and to determine the intrinsic background rate of the spacecraft. Further information, also on the average energy of solar neutrons will be gained by comparison of long time averages of the neutron flux at different solar distances.

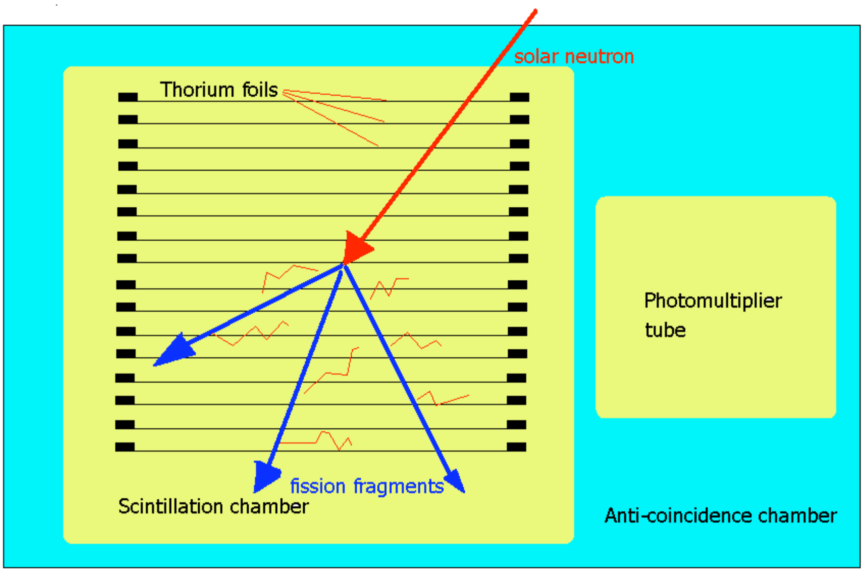


Figure 3.6: Schematics of the NED sensor.

3.8.2 Orbit, Operations and Pointing Requirements

3.8.3 Accommodation

As a resource saving options, this instrument may be co-located together with the GRD instrument.

3.8.4 Interface and Physical Resource Requirements

The main concern with NED will be to minimise the energetic particle background in the sensor cell. On one hand the anti-coincidence cell should be as efficient as possible to prevent false coincidences, on the other hand, the location of the sensor within the payload has to be chosen such that as to keep the particle radiation level to a minimum.

INSTRUMENT SUMMARY

Instrument	Mass [kg]	Power [W]	Volume [cm^3]	Data rate [b/s]
NED	3.5	3	3375	160

Table 3.13: NED resource requirements.

3.8.5 Cleanliness, Ground Operations and Other Requirements

3.8.6 Open Points and Critical Issues

Name / acronym	**Neutron Detector (NED)**

Objectives	1) to provide first measurements of low-energy solar neutrons
	2) to determine existence of a quasi-steady-state neutron halo around the sun
	3) to help elucidate solar eruptive events (e.g. CMEs, flares) and understand the sun as a prolific and variable particle accelerator.

General description	A non-imaging detector measuring fast neutron fluxes in the energy range from ~ 1 to at least 10 MeV. A number of different sensor types may be capable of performing the desired measurements.

Reference P/L and/or heritage	Neutron detectors on Mars Observer, Lunar Prospector, Mars Odyssey,
	Pioneers 10 and 11, various military.

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Fission chamber, scintillator, other. Also need PMT or PD for anti-coincidence field	A number of different technologies may be proposed.
Spectral range	MeV	~0.7 to 10	Upper energy limit may be higher, depending on sensor type.
Operating T	C	-20 to 30 C	Not critical
Optics /antennas			
Type	N/A		
FOV	Sr.	4π	
Bandpass	N/A	None	Responsive to all energies
Pointing	N/A	On nadir (+X) face of spacecraft	Details not critical
Configuration			
Physical Units	No	Single box	
Layout	cm.	~15 x 15 x 15	Size will depend on sensor type
Location S/C	N/A	On shadowed equipment deck (+X). Would prefer location on shadowed boom, if available	Prefer away from large quantities of hydrogen containing materials (e.g. fuel)
Physical			
Mass, total	kg	2 to 3.5 for standalone unit	Mass will depend on sensor type and degree of subsystem sharing
Dimensions	cm	~15 x 15 x 15	Size will depend on sensor type and degree of subsystem sharing. Cubic shape not likely
Volume	Liter	3 to 5	See above
Power			
Average	W	1.5 to 3	Power will depend on sensor type and degree of subsystem sharing.
Peak power	W	1.5 to 3	Average is same as peak
Stand-by	W	1.5 to 3	Some sensor types may have reduced power in stand-by, others not.
Data rate / volume			
Average data rate	Bits/sec	<160	
Peak data rate	Bits/sec	<160	Average is same as peak
Data volume /orbit	Mbyte	<250 for 149 day orbital period	Data rate constant over orbit

Own data storage	MByte	None	Negotiable with spacecraft
Thermal			
Heat load to radiator	W	None	
Operating T range	C	-20 to +30	Not critical
Other requirements	N/A	None special	
Cleanliness			
EMC requirements	N/A	None special	
DC magnetic	N/A	No strong B-field near unit	B-filed could affect PMT, if used.
Particulate	N/A	None special	
Miscellaneous			
Mechanisms	No.	None	No deployables or expendables
Alignment		Not critical	
Orbit requirements		None special	Nadir (+X) to Sun
AIT/AIV requirements		May require radioactive source for testing / calibration	No purge required

Development approach / schedule

Preferred model philosophy	Provide EM and FM. Project determines need for other models
Estimated development time	EM (1.5 yr), FM (1 yr)

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
None identified	All pertinent technologies fairly well in hand. Modeling of specific sensors required

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
ND		5	Detailed designs could be generated fairly quickly for most sensors types that may be proposed.
ND	7-9		The detector technologies are fairly mature and have been flown in space in various incarnations but the specific systems that will be propose for SO have not flown and will be new designs

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

4. Solar remote-sensing instruments

4.1 Visible-Light Imager and Magnetograph (VIM)

4.1.1 INSTRUMENT DESCRIPTION

SCIENTIFIC DRIVERS

The purpose of the VIM is to measure the magnetic and velocity fields in the photosphere. It provides the magnetic boundary for the MHD processes observed by other remote sensing instruments and allows surface and subsurface dynamics and structure to be determined, e.g., with the methods of local helioseismology. It will observe the morphology, dynamics, and strength of the magnetic elements and flux tubes at the photospheric level with a resolution that is consistent with the resolution of the EUV telescopes. It will also provide the first images, Dopplergrams and magnetograms of the solar poles and of the side of the Sun, which is invisible from Earth.

VIM will have vector magnetic field capabilities as this is of fundamental importance to understand the nature of photospheric fields: Are the polar fields vertical unipolar fields? Do they harbour complex neutral lines with horizontal sheared fields? Having vector capabilities is also the only way in which quantitative inferences of the magnetic field in the transition region and corona can be made (from force-free or full 3D MHD extrapolations).

VIM will also produce line-of-sight velocity maps by observing two points on either side of a spectral line. These maps can be used, through local helioseismology techniques, to investigate subsurface flows. The internal structure and dynamics of the near-polar regions of the Sun is of paramount importance and perhaps *the* key to our understanding of the solar cycle.

The key scientific goals of the VIM are

- to provide measurements of the "magnetic carpet" which drives chromospheric and coronal activity as studied by the UV and X-ray instruments.
- to provide surface and subsurface flows in the field of view of the UV and X-ray instruments.
- to observe and accurately quantify for the first time the surface polar magnetic field of the Sun.
- to measure rotation and flows near the Sun's poles using techniques of local area helio-seismology and thereby provide crucial constraints on solar dynamo theories.
- to unveil the small-scale photospheric dynamo.
- to resolve solar magnetism down to its fundamental length scale (<100 km).
- to provide the first magnetograms and Dopplergrams of the far side of the Sun.

INSTRUMENT CONCEPT

The optical concept of VIM includes

- 25 cm diameter High Resolution Telescope (HRT),
- 2 cm diameter Full Disc Telescope (FDT), and
- Filtergraph Optics (FO).

The two telescopes, HRT and FDT, deliver an image of the Sun at a common telescope focal plane. The FO produces a magnified image of the telescope focus on the instrument's single detector. While the HRT provides an image with a high resolution at its diffraction limit, the FDT sets its focal length to fill the detector with the solar disc at 0.21 AU. The filtergraph optics include a double Fabry-Perot interferometer equipped with a narrow-band interference filter as a prefilter, tuned to an Fe I line in the 500 nm region. This filtergraph is easily adjusted to the lineshifts which occur due to Sun-spacecraft relative velocities up to 40 km s^{-1}. The three components are described in the following sections.

High Resolution Telescope (HRT)

Two solutions are being evaluated for the HRT, first, an open solution with C/SiC mirrors (from the assement study) and a heat stop rejecting most of the solar light outside of the spacecraft and, second, a closed solution with a window (possibly a lens) coated to allow a very small wavelength region entering the telescope.

- *General description: open solution*

The HRT is a 250 mm diameter Gregorian reflector. It provides an image with a spatial resolution as high as 75 km on the solar surface during the near-Sun orbital phase. A polarisation modulator is foreseen in the optical path behind the main mirror in the f/17 beam and before the first oblique reflection.

The secondary mirror is rigidly mounted and aligned to the main mirror to work in a well defined temperature regime.

Figure 4.1 shows an annotated perspective view of the HRT and FO.

- *Thermal control: open solution*

The HRT will have to withstand the heat load during the near-Sun orbital phases. The primary mirror will be made from C/SiC and cooled with a dedicated radiator to dispose off the approximately 170 W of radiation which is absorbed in its reflective layer. A switchable heat-pipe will regulate the temperature of the mirror to a nominal value. C/SiC enjoys better static and dynamic thermal stability than other mirror blank materials. The structure of the telescope will also be made out of the same material.

The Gregorian design permits rejecting, at least, 96% of the incident radiation at its primary focus with a field stop (heat stop). The field stop is an inclined mirror with a 2 mm central hole which transmits a field of view of 14 arcmin. The reflected light is directed out of the spacecraft. The field stop will also absorb about 170 W which will require a second dedicated radiator.

A broadband interference filter will be included after the field stop to reduce the amount of light in the filtergraph optics.

- *General description: closed solution*

To avoid high thermal loads on the telescope primary mirror a closed HRT telescope is currently investigated.

The conceptual design is a 250mm aperture Cassegrain telescope with a parabolic primary mirror of 500 mm focal length made from lightweight C/SiC technology. Together with the hyperbolic secondary the effective focal length of the telescope is 8500 mm, resulting in an f-number of 17. The entrance aperture of 250 mm is covered with an entrance window of Suprasil (fused silica), which has been covered on the backside (inside VIM and, thus, protected) with a multilayer reflective coating. The entrance window can be either a monolitic Suprasil plate with 20mm thickness, polished to a wavefront distortion of □/20, or a segmented window with 6 round subapertures of 80mm each with a thickness of 8mm. This accounts for the central obstruction of the secondary mirror, which is 80mm in diameter.

The multilayer coating is designed to provide maximum overall reflected energy. Only in a 50-100 Å wide bandpass centred around the observing wavelength, the transmission of the filter will be high. Together with the transmission properties of fused silica the filter will provide blocking of UV light below 360nm, in order to avoid polymerisation of organic substances on optical surfaces. The identical multilayer design is also used for the full-disk telescope. In the infrared part, complete absorption is provided by the Suprasil substrate. The front face of the window is an effective thermal radiator and radiates a large part of the absorbed energy back into space, the rest is transferred to the mounting and a conventional heat-pipe/radiator assembly. This ensures that the filter temperature is not exceeding 200 degrees C at any time of the orbit.

The use of the monolithic entrance window itself as the objective lens of a refractor type HRT telescope is also investigated. As VIM is a monochromatic instrument this lens would be a simple singlet optical component.

- *Thermal control: closed solution*

The only critical element from a thermal point of view is the entrance filter (or lens). Assuming a broad band absorption of 10 %, 170 W will have to be disposed off by a dedicated radiator. The main saving compared to the open case is the reduction of critical elements from two (mirror+heat rejection) to one (entrance filter).

Full Disc Telescope (FDT)

It is impractical to combine the functions of full-disc and high resolution viewing of the Sun into a single telescope. Therefore, a separate telescope will provide a full disc image at the same position as the secondary image of the HRT. The filtergraph is therefore shared by both telescopes. Light from either the HRT or the FDT will be selected by a shutter mechanism. The FDT is composed of a coated first lens and a relay system, both chosen to provide a full disk image at perihelion distance that fills the detector with the image and working at the diffraction limit. The diameter of the FDT is such that the ratio of the apertures between the HRT and the FDT is equal to the inverse of the ratio of the field-of-view of both telescopes. This ensures the same field-of-view performance for both telescopes of the FO Fabry-Perots.

A cube beam-splitter sends a small fraction of the light of the FDT to a limb-sensor that includes a folding mirror acting as closed-loop tip tilt system to stabilize the image to 0.01 arcsec during an interval of 10 seconds (typical integration time). This Image Stabilization Sytem (ISS) will always work, even if VIM is observing with the HRT. It will derive the correction signal needed to compensate spacecraft pointing errors and will send the correction signal to similar tip-tilt mirrors in the HRT and other remote sensing instruments (those needing a better pointing accuracy than that provided by the AOCS of the spacecraft). A calibration strategy of all these tip-tilt mirrors with respect to the one on the FDT path needs to be defined to ensure a correct performance during the mission lifetime.

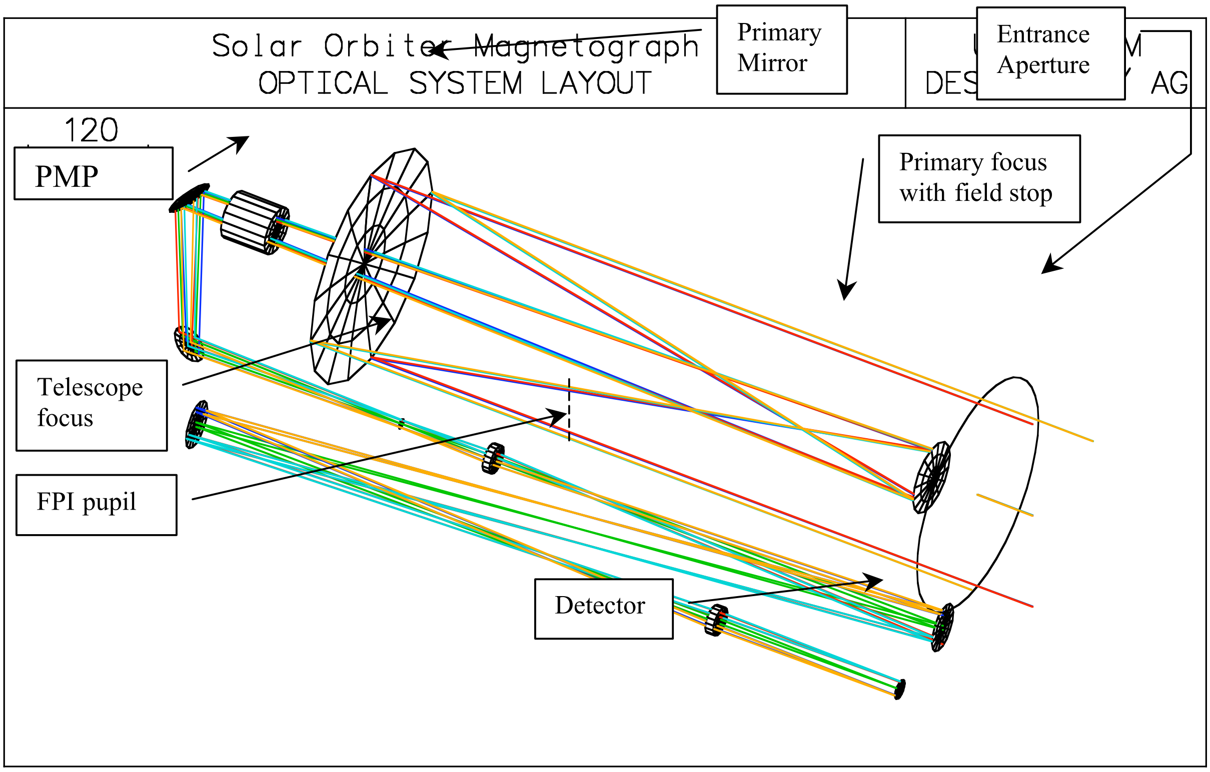


Figure 4.1: Perspective view of the optical layout of the high resolution telescope and filtergraph.

The FDT shares the control electronics with the HRT.

A coating at the entrance lens reflects most of the incident sunlight so that the radiation load is not a problem for the FDT.

Figure 4.2 shows the annotated optical layout of the FDT from a similar point of view as the HRT in Figure. 4.1 (the ISS is not shown). Note that the layout is "horizontal" (in the XY plane) compared to the vertical (YZ plane) layout of the HRT proper (see Fig 4.4). A separate PMP is necessary for the FDT in order to perform polarisation modulation within the centred optical path before the first oblique reflection.

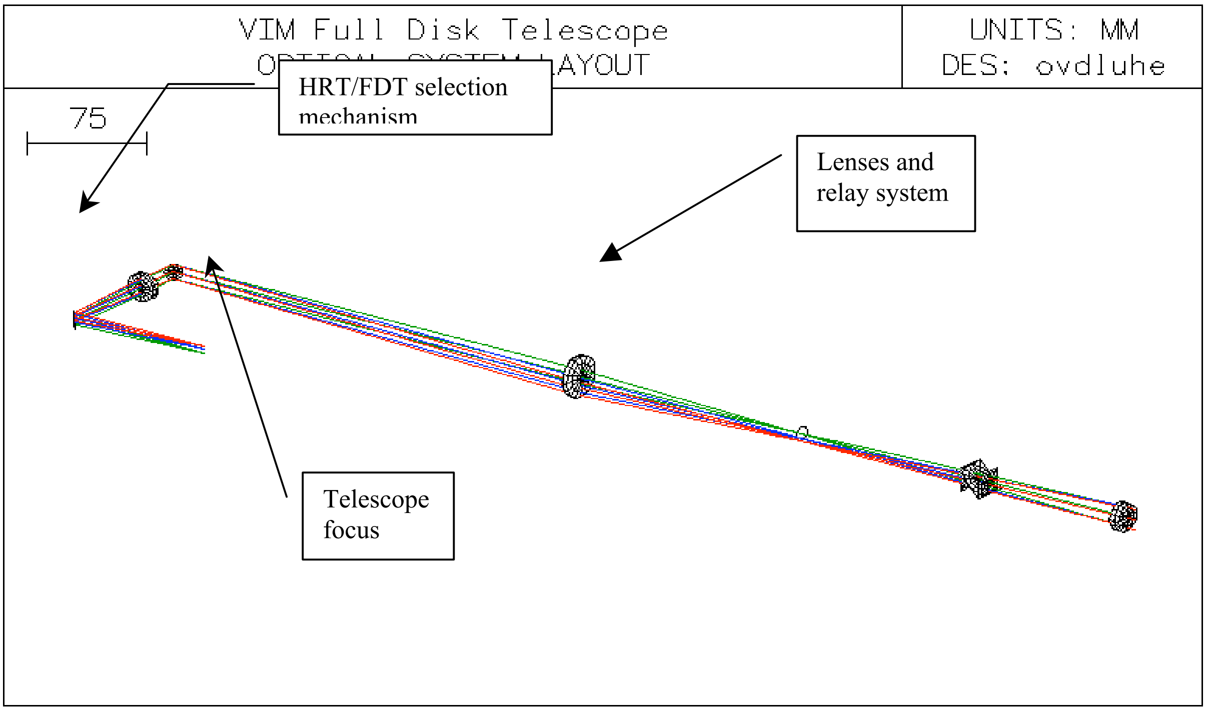


Figure 4.2: Perspective view of the optical layout of the full disc telescope.

Filtergraph Optics (FO)

The FO consists of two singlet lenses which combine the functions of collimator and camera with a 1:2 magnification. Two flat mirrors are used to fold the beam and provide room for a collimated light path with a pupil of 44 mm size available to locate the Fabry-Perots. The maximum entrance angle at the Fabry-Perots is 0.58 degrees, which produces a wavelength shift of 50 mÅ from the centre to the corner of the detector. The total optical path length of the FO is about 3.5 m long. A cold stop near the pupil prevents the filters and detectors from receiving radiation from the hot HRT telescope.

Two 50 mm diameter Fabry-Perot etalons near the pupil plane provide the required spectral resolution of typically 50 mÅ. One etalon will provide the spectral resolution while the other blocks the secondary transmission maxima of the first. A separate interference filter with a 3 Å band blocks the secondary transmission maxima of the combined etalons.

We consider using two $LiNbO_3$ solid state etalons with fixed resonator widths which are mounted on a temperature controlled oven (0.1 degrees of stability is required for 10 mÅ passband shift). Spectral tuning is achieved by applying voltages to the Fabry-Perots. About ±2 kV is required for a shift of the passband of ±1 Å, which is sufficient to cover both line width and Sun-spacecraft velocity shifts. The $LiNbO_3$ technology will require a thorough space qualification effort, with particular emphasis on the performance under high particle radiation environments. This technology has as main advantage the lack of moving parts while it allows to achieve finesse values as high as 30. The fast tunability times (better than ms) and the high refractive index of the material (n=2.3, which simplifies field-of-view problems) are two other interesting properties of this type of etalons. $LiNbO_3$ Fabry-Perots have been used successfully in stratospheric balloon experiments (Flare Genesis). As back-up technologies, one could use PZT spacing controlled Fabry-Perots with flight heritage from the HRDI instrument in the UARS satellite (requiring moving parts) or liquid crystal Fabry-Perots that are under development (in the US) for Earth observing missions.

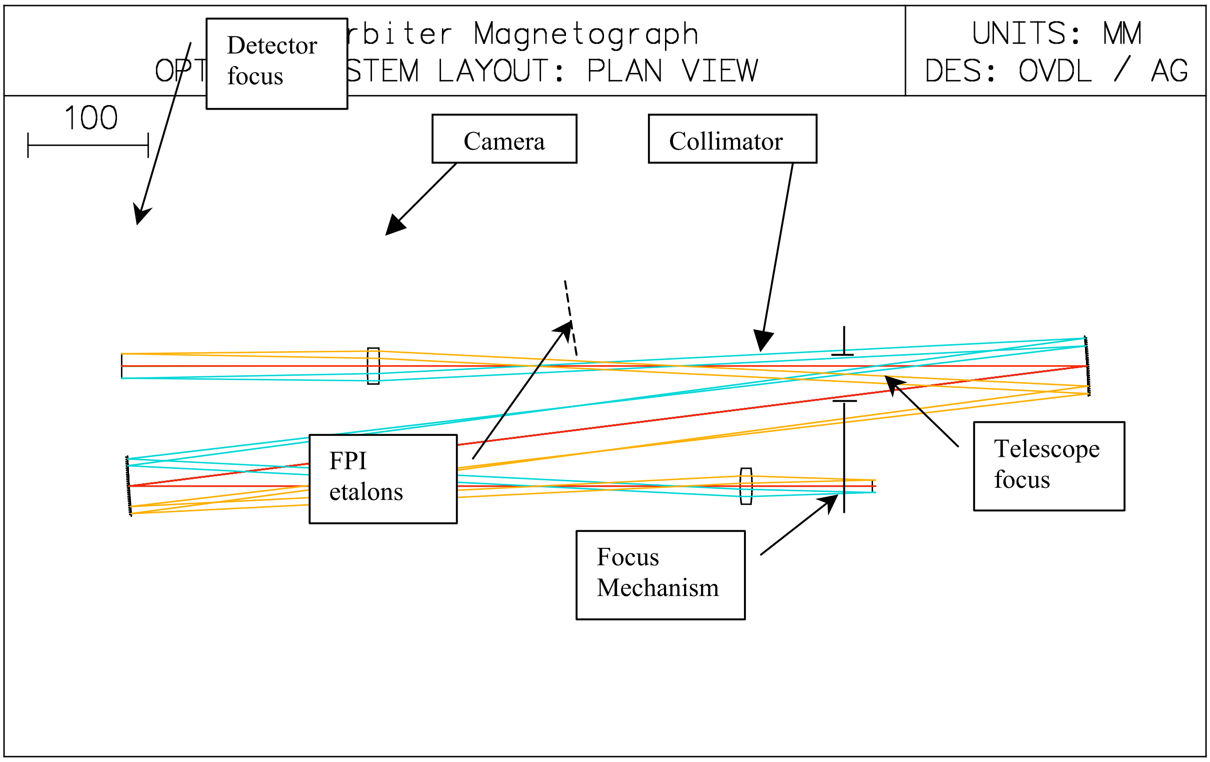


Figure 4.3: Arrangement of filtergraph optics.

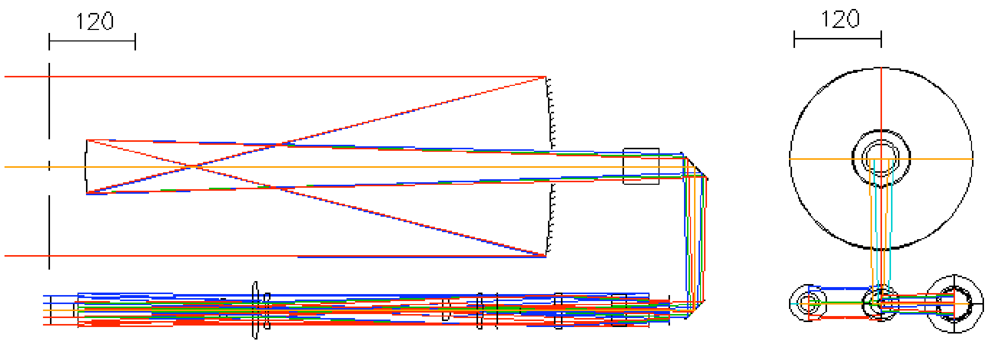
A focus mechanism near the telescope focal plane is used for accurate focusing and to re-image the pupil onto the detector (MDI heritage). This last calibration mode will allow a pixel-to-pixel calibration strategy of VIM for wavelength registration.

The FO is mounted together with the FDT on a common optical bench.

Figure 4.3 shows a plane view of the filtergraph optics. The etalons are not included but their position is indicated.

Overall configuration

The combination of all optical subsystems is shown in Figure 4.4. The HRT is the large cylinder on top of the Y and Z views. The FDT is to the lower left of the Z view and near the bottom of the X view. The FO are to the lower left of the Z view and near the top of the X view.



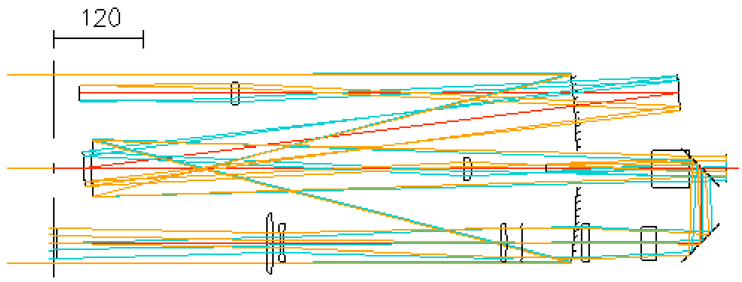


Figure 4.4: Plain views (top left: Y, top right: Z, bottom: X) of the combined VIM. The HRT is arranged above the FDT and the FO, both of which are mounted on a common optical bench. The optics fit a 300 mm x 400 mm x 1300 mm envelope.

Polarisation Modulation Package (PMP)

The PMP will allow VIM to provide longitudinal and transverse magnetograms of the region being observed. The PMP will produce the modulation of the intensity at the APS detector as a function of the input polarisation state. These intensity changes of the APS detector measurements will be used to recover the Stokes vector of the solar light.

VIM will use two PMPs, one for the HRT and one for the FDT. Each PMP will be composed of a couple of Liquid Crystal Variable Retarders (LCVRs) followed by a fixed linear polariser. The LCVR retarders follows the design of ground polarimeters successfully built and used at the Canary Island Observatories. LCVRs produce polarization modulation using simple square waves with amplitudes of up to ±20 V. They need to be temperature controlled to within 1 degree. LCVRs can be built in such a way that, for no applied voltage, no net retardance is introduced (compensated LCVRs). In this case no effect is produced when only velocity measurements are being made. The LCVRs combination generates 4 independent polarization states that are read by the APS detector. LCVRs have been tested to some extent for space applications but a full characterization for the Solar Orbiter environment is needed, in particular sensitivity to UV light and continued performance under vacuum conditions. LCVRs will be used in the Spanish-led Imaging Magnetograph experiment, IMaX, for the SUNRISE stratospheric balloon project (within the NASA-LDB program) and led by MPAe in Germany.

APS detector cameras

VIM uses only one detector at the focal plane of the FO. The detectors is baselined to be a CMOS-APS 2048 x 2048 pixels detector. In the following table we describe the specifications of the detector for VIM and we compared it with existing CMOS-APS detectors. For the light flux budget, we are assuming here 0.25 arcsec per pixel, a telescope working at diffraction limit (so telescope size plays no role), 50 mÅ spectral resolution and a total transmission of 6 % of the system telescope+filtergraph. For the desired sensitivity of 4 G in longitudinal magnetic field (10^{-3} in polarization accuracy), a detector with the following specs is needed:

Description	**Ideal case**	**Ibis5**	**ACS-I2048**	**Star1000**
Array Format	2048 x2048	1280x1024	2048x2048	1024x1024
Pixel size (μm)	< 10	6.7	12	15
Quantum eff.(%)	> 85	50	50	45
Fill factor (%)	100	60	60	45
Readout noise (e^-)	<150	50	50	54
Dark current (e^-/s)	<1000	750	1200	6000
Full well (e^-)	200000	60000	600000	160000
Frames per second	<12	1-30	<6	<12
Power consumption (mW)	< 500	350	150	400

Operating Temp. (^oC)	-5 to 20	22	0 to 70	21
Shutter capacity	Snapshot	Snapshot	Rolling	Rolling
ADC (bit)	10-12	10	10	10
Radiation tolerance (Krad)	>100	Unknown	Unknown	250
FPN/PRNU (%, rms)	<1	0.5/2	0.5	
DCNU (%)	<5	15		
Optimun fps	12	12	1	3
Detected e^-	140612	41357	496279	111663
S/N of single frames	348	197	702	329
Exposure for S/N=1000 (s)	3.2	8.7	8	12
Provider	C3PO Development	Fillfactory Existing	PVS Existing	Fillfactory Existing

Table 4.1: Detector requirement specifications and comparison with presently existing cameras.

The second column of table 4.1 provides the required specs for the VIM detector and the last three provide the performance of existing devices. The Ibis 5, from Fillfactory, has as main drawbacks the small array size (but Fillfactory has already developed a 2K x 2K device, column 2 shows an already existing APS with this size) and a low quantum efficiency-filling factor product. These small values force the total integration time per wavelength position to be of 8 seconds which is the upper limit of what will be admissible (a sound wave takes 6 seconds to cross one pixel and twice that for the resolution element). A radiation tested detector (like the Star1000 from the same company) with the required array dimension and some improvement on photon detection capabilities would be a feasible solution for VIM, requiring minimum development on the proposing team sides. This is the nominal case used for the mass and power breakdown. Note that the operating temperature of these detectors is near room temperature values.

As a major detector development that could reduce the total exposure time (by improving the product QE*fill factor) to values near the specified 3.2, we are studying the development of the C3PO (Charge Caching CMOS detector for Polarimetry) detector that also offers other advantages. C3PO uses the hybrid CMOS silicon deposited technology (developed by Rockwell Scientific, USA). It shares the properties of backside illuminated CCDs (100 % filling factor and high quantum efficiency) with the flexibility, low power consumption and mass requirements of CMOS devices. In addition, it includes, per pixel, a total of 4 integrating capacities that hold the charge from the current exposure and 4 read-out capacities that can be used to integrate different exposures on-chip until the required S/N is achieved. Only then, the detector is read. A C3PO device like this is going to be developed by a European consortium for ground applications (using large pixels). A similar development (with ESA support) for smaller pixels adequate for VIM could be undertaken. We want to stress that the advantages of this development are, not only the reduced exposure times, but also that it requires a slow framegrabber without the FPGA or ASIC charge integration circuitry that would be needed for the existing detectors.

4.1.2 Orbit, Operations and Pointing Requirements

In general, we envisage practicing encounter periods before the real event takes place and after the target and science objectives have been selected for each perihelion pass. VIM operations should be considered in parallel with the rest of the remote sensing observations.

HRT and FDT cannot be used simultaneously as they share the same FO. They will be used sequentially during the 30 days of nominal observations during the co-rotation phases and the out of the ecliptic periods. Occasionally FDT could be used for observing the back side of the Sun and as a back-up to other programs that could be considered.

Outside of the nominal 30 encounter days, a calibration mode of the tip-tilt mirrors in the FDT and HRT path should be programmed (preferably before the encounter phase and while the spacecraft has direct contact to the Earth). This calibration program (that should be considered by all instruments receiving the

VIM stabilisation signal) will allow to set the gains and offsets of the PZT normally included in the tip-tilt mirrors and that, will inevitably, suffer from degradation during the mission lifetime.

The generation of spurious (velocity and magnetic field) signals by spacecraft jittering requires the use of the ISS installed in the FDT path. The pointing requirement set by the scientific requirements is 0.01 arcsec (1) in 10 seconds which sets the requirement to 0.001 arcsec/s. Depending on the final spec set by the frame rate of the detector used, this requirement could be relaxed up to 0.01 arcse/s. Existing experience with limb sensors on-board the SOHO and TRACE spacecrafts shows that these requirements can be reached with the system proposed here (FDT-ISS concept).

VIM is hard-mounted on the spacecraft and aligned to within 2 arcmin to the other instruments.

4.1.3 ACCOMMODATION

The large aperture of the HRT-VIM assures a spatial resolution matching that achieved with the UV instruments of the Solar Orbiter. But it also makes the thermal balance of the instrument most demanding. We anticipate the need of radiators filling one side of the instrument (1.1 x 0.4 m^2) and facing deep space. As a result VIM should be located on the periphery of the spacecraft payload allocation.

The open solution of the HRT has an extra requisite on the spacecraft as it rejects a significant amount of light on one side of the instrument. This outgoing beam should be directed towards an opening not interfering with any other instrument or the spacecraft itself.

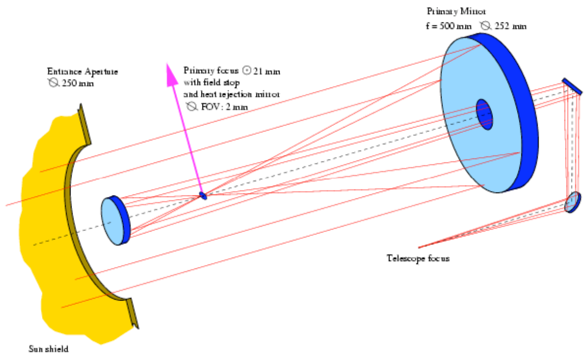


Figure 4.5: Schematic diagram of the HRT and the light rejection system at the primary focus field stop.

VIM entrance aperture will be crossed by the telescope spider and secondary mirror. VIM assumes the spacecraft sunshield protects all these components so that they are not directly illuminated by solar light. An aperture door mechanism will fit between the sun shield and the telescope entrance. The door will open sideways, moving behind the sun shield.

4.1.4 INTERFACE AND PHYSICAL RESOURCE REQUIREMENTS

Telemetry needs – data compression.

VIM will detect intensity images in different positions within a selected spectral line and in different polarization modes (including an unpolarized mode). For calibration purposes, sometimes, these intensity frames (or the Stokes parameters easily deduced from them) will be stored. But these data will represent a small fraction of the total and will not compromise the telemetry rates. Here we consider only the cadences and telemetry rates needed for different observing modes that should constitute the fundamental science operation modes of the instrument. The use of these modes will depend on the science targets selected for each orbit based on the science plans of the spacecraft. In any of these modes, VIM will provide a combination of the following physical magnitudes:

1. Ic or continuum intensity images. A temperature indicator that provides the photospheric context. 8 bits compressed to 4 bits per pixel.
2. V_{los} the line-of-sight (LOS) velocity frames. They provide the Doppler signals needed for local helioseismology. 10 bits compressed to 5, some applications may use only 4.
3. B_{los} the LOS component of the magnetic field. They are basically maps of circular polarization over the observed area. 10 bits compressed to 5, some applications may use only 4.
4. B_{trans} the transverse to the LOS component of the magnetic field. They represent maps of linear polarization. 8 bits compressed to 4.
5. the azimuth of the transverse component in a plane perpendicular to the LOS. Also obtained from linear polarization measurements. 8 bits compressed to 4.

The final 4/5 bits per pixels estimates provided here, assume a lossless compression scheme with an efficiency of a factor 2. Note that from the original 12 bits, we have first thrown out the 2 to 4 less significant ones. Thus the total reduction factors are between 2 to 3. These compressed estimates have been used in the following description of example observing modes that could produce the desired scientific results from VIM:

Mode 1. Low resolution, high cadence mode: On-chip binning to 512 x 512 pixels of 1 physical magnitude at a cadence of 1 per minute require a telemetry rate of 18-22 kbs. This mode can be used for storing V_{los} over the whole FOV at a high cadence adequate for local helioseismology.

Mode 2. Medium resolution, medium cadence mode: Binning to 1024 x 1024 pixels of 1 physical magnitude at a cadence of 1 every two minutes require a telemetry rate of 35-44 kbs. This mode can be used for sending Ic, V_{los} or B_{los} for general purposes (e.g., magnetic field evolution).

Mode 3. High resolution, high/medium cadence mode: This mode is similar to the two previous ones but instead of binning pixels, a selection of a subframe (512 or 1024) is done, thus prioritizing spatial resolution at the expenses of FOV and keeping a reasonable cadence.

Mode 4. Photospheric context: In this mode three quantities (B_{los}, B_{trans} and for vector magnetometry or B_{los}, V_{los}, Ic for dynamical studies) can be sent over the full 2K frame every 5 minutes at a rate of 160 kbs (4 bits per magnitude). The vector magnetometry case enables to follow the evolution of the magnetic field at a sufficiently high cadence adequate for most of the upper atmospheric phenomena.

Modes 1 and 3 have telemetry rates similar to the nominal one 20 kbs. The others exceed this value by a factor 2 to 8 . Peak telemetry rates of 3 physical magnitudes over the full frame every minute of 800 kbs must also be considered (with a low duty cycle).

As it is evident VIM will require often larger rates than the nominal one. The way in which this could be achieved is:

- By using those orbits where the telemetry rates of the spacecraft are larger than the nominal 75 kbs, by factors between 2 to 8. Thus careful planning of the scientific objectives for each orbit should be made beforehand.
- By increasing the resources of the spacecraft (in realistic ways as increased on-board memory, a second downlink ground station).

- By studying lossy approaches that do not compromise some of the science objectives. This point should be of interest to all instruments and experience from other spacecrafts under development is available. We propose to study this point in a coordinated way between all science teams in the future.
- By pre-programming flexible data acquisition rates in parallel with the rest of the instruments.

As it is shown, with the nominal telemetry rates VIM will be able to provide at all orbits the magnetic and velocity context in which the upper atmospheric processes studied by the rest of the instruments take place. However, the nominal 20 kbs is only marginally satisfactory for this purposes and it would exclude the fastest evolving solar processes from being studied. An increase of the nominal telemetry by a factor 2 to 8 is strongly recommended. The science goal of local helioseismology puts a high demand on observing cadence but not so strong on spatial resolution. Modes 1 and 2 can be accommodated in different orbits to achieve this science goal.

Mass breakdown and power requirements

	Mass (g)
1. High Res. Telescope (incl. FO, APS)	**10100**
- 25 cm telescope (mirrors + structure)	2800
- Prefilter	100
- PMP	400
- HRT folding mirror	100
- Tip-tilt mirror	800
- Re-imaging system (4x100)	400
- VIM focus mechanism	500
- HRT/FDT selector	300
- Fabry-Perot, Oven and electronics	2500
- APS detector camera + electronics	1200
- Harness	1000
2. Full Disc Telescope (incl. ISS)	**3100**
- Prefilter	100
- Lenses and structure	600
- PMP	400
- FDT folding mirror	100
- Tip-tilt mirror	800
- Cube beam-splitter	200
- ISS lens	100
- ISS detector and electronics	500
- Harness	300
3. VIM enclosures	**6500**
4. VIM thermal subsytem	**3500**
5. VIM electronics (DPU and control electron.)	**3000**
6. VIM door	**800**
7. VIM margin	**3000**
8. VIM TOTAL	**30000**

Table 4.2: VIM mass break-down.

	Power (W)
APS+electronics	4

Image Stabilisation System	4
Fabry Perot & Oven	8
PMP	2
DPU and control electron.	5
Margin	2
TOTAL	**25**

Table 4.3: VIM power requirements.

4.1.5 Cleanliness, Ground Operations and Other Requirements

Cleanliness requirements.

The varying thermal environment, as well as the changing apparent size of the Sun, increases the risk of redistribution of outgassing contaminants during the orbit. The deposition of these species on the optical components is extremely enhanced on cold surfaces and on surfaces exposed to solar UV light and particle flux, leading to irreversible deposition by polymerization of the organic substances.

A filter will block the ultraviolet component of the spectrum on a clean, hot surface very early in the optical path. The optical path before this filter will be extremely clean and free of outgassing organic material. The UV filter will block all wavelengths shorter than 360 nm. Since the working passband of the instrument is in the visible, a UV blocking filter at the entrance aperture would be a preferred solution from a cleanliness point of view. A hot telescope with a filter behind the secondary may also be an acceptable solution. In any case, the instrument must be ultimately clean up to this surface, like a solar UV instrument. However, the filter must be stable against the radiative flux and must be unpolarizing. These two requirements must be studied before this can be considered a final solution.

During ground operations the HRT will be closed by an openable cover (door mechanism) that will allow purging with clean gas. To avoid contamination build-up during operation in orbit, the thermal design of the VIM instruments will ensure that there are no optical surfaces colder than their surroundings.

Component Requirements and Precautions against Latch-up and Single Event Upset.

Electronic parts used in VIM shall be insensitive to SEL (Single Event Latch-up). The Linear Energy Transfer (LET) threshold for SEL is estimated to be > 100 MeV cm_/mg. If it would be necessary to use parts with a lower LET threshold, current monitoring and over current switch-off will be introduced to avoid single event burnout.

The component's insensitivity concerning Single Event Upset (SEU) shall have an estimated LET level of more than 40 MeV cm_/mg. In addition to this selection a Single bit Error Correction and Double bit Error Detection (SECDED) unit will protect program and data memory against errors caused by SEU. If double bit errors or other malfunctions in the central processing unit occur, a hardwired watchdog circuitry will initiate a system reboot after the software fails to send the cyclic trigger signal. This strategy will provide automatic detection of SEUs and resumption of full operability, to ensure no disruption in the science operations during phases when no NRT communication with the instrument is possible.

Ground operations.

The VIM instrument will be operating by execution of predefined observation sequences to be stored in the DPU. Execution of sequences can be invoked by time tags. In this way, a pre-planned, long-period observational phase can be started and carried through an encounter phase. Such pre-planned observations must be practiced by the instrument prior to the encounter. The VIM instrument will need a dedicated Ground Support Equipment (GSE) to build these observational programmes and transfer them via the spacecraft.

Commanding capability must be provided by the project for the operation of mechanisms (aperture door, focussing mechanism) and for adjustments of the thermal configuration of the instrument.

4.1.6 Open Points and Critical Issues

1.- Thermal concept: Both open and closed solutions need a feasibility study providing the temperatures at which the HRT should be working, the impact on the optical quality of the varying thermal loads and the resources needed by the thermal subsystem. Only the basic considerations have been made so far with simple thermal models. They have suggested that for both solutions some elements or the entire main telescope should be working at a nominal temperature in the range of 150-250 ^{o}C. The mass assigned to the radiator and heat pipes controlling this thermal environment has been estimated to be of 5 Kg, with the closed solution being more realistically close to this number. We note that the visible APS detector is baselined to work at a temperature no cooler than -5 ^{o}C, which will probably easily achievable using passive modes.

Industrial studies focusing on the thermal concept for both solutions are under way and will be finished by mid February 2003. These will be the basis for very detailed thermal model calculations needed to accomplish stable thermal conditions in orbit.

2.- Polishing of ultra-light wave mirrors: Materials, like C/SiC, can produce mirrors with areal density smaller than 10 Kg/m^2 making them extremely suitable for applications like the one described here. In addition this material has a very large thermal conductivity and can be used much hotter than other materials. Thus producing a mirror with a substrate these materials is potentially very important for VIM.

3.- CMOS detector: Available APS-CMOS devices can be considered to be a good starting point to develop the detector needed for VIM. The only area that needs further development is the increase of the typical QE times filling factor properties of them and their radiation tolerance. One way to reach this performance would be the development of the C3PO detector for space applications in parallel with its ground version.

4.- Compensated LCVRs and their space qualification: These devices offer a light and low power solution for the VIM PMP. They have been successfully used for ground-based instruments and tested to some degree for space applications. Uncompensated prototypes have been produced by a collaboration between IAC and an LCD company (in Spain) for use in a balloon experiment. Full space studies for the Solar Orbiter case are needed. In particular a careful calibration of their malfunctioning under UV light is needed.

5.- $LiNbO_3$ etalons: this technology has been used in a variety of instruments for ground applications.
MPAe (Lindau) is studying their performance for space applications. One aspect that needs careful characterization is how they behave in radiation harsh environments when kilo-volts driving signals are being used.

6.- Multilayer coatings: Stability of the coatings for the interference filters and mirrors under high thermal load.

Name / acronym	**Visible-light Imaging Magnetograph / VIM**

Objectives	
	1) to detect solar surface magnetic fields and velocity flows.
	2) to provide the photospheric context for the rest of the Solar Orbiter instruments
	3) to probe polar subsurface layers using helioseismology

General description	Imaging spectroscopy with polarization capabilities. The design uses a tunable filter to select a specific wavelength band within a spectral line and a polarization modulation system. It provides image stabilization error signals to other imaging instruments in the S/C.

Reference P/L and/or heritage	
	e.g. SOHO MDI, Solar-B FPP, SDO HMI
	e.g. DLR+NASA Sunrise balloon experiment carries similar instrument being developed by a Spanish consortium.
	e.g. ample space (mostly US) and ground (European and US) experience in similar instruments.

Parameter	**Units**	**Value / Description**	**Remarks**
Sensor / detector			
Type	1	CMOS APS	2K x 2K (may be larger)
Spectral range	Å	One spectral region in the range 4000- 7000	Optical design is monochromatic.
Operating T	C	-5	
Optics / antennas			
Type	N/A	Reflect, refract, filter optics, polarization optics	High res. telescope + Full disk telescope (guider) + common focal plane instrument.
FOV	arcmin	high res: 8.5 x 8.5 full disk: 160 x 160	Full disk channel used for limb tracking.
Bandpass	Å	50-100	Prefilter bandpass. Etalon bandpass would be 50 m Å.
Pointing	arcsec/s	0.01/10=0.001	Provided by spacecraft pointing plus image stabilization system.
Configuration			
Physical Units	2	Optical box and electronic box	
Layout	N/A	Close to each other	
Location S/C	N/A	Aperture (25 cm maximum) on the front shield. Side location with access to space for radiators and light rejection system.	Light rejection system is TBC.
Physical			
Mass, total	kg	30	Including thermal system
Mass unit 1	kg	23	Telescopes + focal plane instrument
Mass unit 2	kg	7	Electronics
Dimension 1	cm	110 x 40 x 30	Telescopes + focal plane instrument
Dimension 2	cm	20 x 20 x 20	Electronics
Power			
Average	W	25	
Peak power	W	TBD	
Stand-by	W	TBD	
Data rate / volume			
Average data rate	Bits/sec	20000	

Peak data rate	Bits/sec	8 10^5	
Data volume /orbit	KByte	TBD	
Own data storage	MByte	TBD	
Thermal			
Heat load to radiator	W	172	
Operating T range	C	Open telescope: Mirrors and heat stop at 150-200. Optics at 50. Filter solution: filter at 150-200. Optics at 50.	Two possible solutions being considered: an open telescope and a filter bandpass selection at entrance.
Other requirements	N/A	TBD	
Cleanliness			
EMC requirements	N/A	TBD	
DC magnetic	N/A	TBD	
Particulate	N/A	TBD	Open vs. filter solution have a direct impact here.
Miscellaneous			
Mechanisms	5+1	1 aperture door, 1 HRT/FRT selection, 1 focusing system, 1 Image Stabilization System.	Image stabilization might need two tip-tilt mirrors. Secondary deformable (wavefront sensing) is TBC. Polarization modulator is not a mechanism if use is made of liquid crystal devices.
Alignment		Hard mounted on S/C. All instruments co-pointing within 2 arcmin.	
Orbit requirements		rms pointing stability of S/C.	Specs are input to image stabilization system.
AIT/AIV requirements		TBD	

Development approach / schedule

Preferred model philosophy	EM, QM, FM
Estimated development time	EM (1 yr), QM (2 yr), FM (1 yr)

Critical areas: Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
Detectors	7	3	Existing CMOS-APS detectors for space up to 1K x 1K. Low QE*filling factor.
Thermal	1	6	Two possible VIM configurations depending on detailed thermal analysis: open VIM vs. entrance filter VIM. Studies under development.
Polarization Modulator	6	5	Mechanical system vs. liquid crystal based: both would need ample qualification and testing. Mass and power savings favor a liquid crystal solution.

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):

1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

4.2 EUV Imager and Spectrometer (EUS)

4.2.1 Instrument Description

SCIENTIFIC DRIVERS

Spectroscopic observations of emission lines in the UV/EUV region of the electromagnetic spectrum are critical for the determination of plasma diagnostics from the solar atmosphere, providing the necessary tools for probing the wide solar plasma temperature range, from tens of thousands to several million K. Analysis of the emission lines, mainly from trace elements in the Sun's atmosphere, provides information on plasma density, temperature, element/ion abundances, flow speeds and the structure and evolution of atmospheric phenomena. Such information provides a foundation for understanding the physics behind a large range of solar phenomena.

Current spacecraft instrumentation (SOHO) provides EUV spatial and spectral resolving elements of order 2-3 arcsec and 0.01 nm, respectively, and UV resolutions of 1 arcsec and 0.002 nm. There is no EUV or UV spectroscopic capability aboard the NASA STEREO mission (2005 launch), the NASA Solar Dynamics Observatory (2007 launch) and the NASA Solar Probe (currently in a study phase). The only planned EUV spectrometer for a future solar mission is the EIS instrument on Solar-B with 2 arcsec (1500 km on the Sun) and 0.001 nm resolutions. In addition, the wavelength selection of EIS is tuned to higher temperature, coronal, plasmas, particularly well suited to studies of active regions and even flares, but with little transition region capability.

The European solar physics community has a well established expertise in solar EUV/UV spectroscopy as illustrated by the successful CDS and SUMER instruments on SOHO.

INSTRUMENT CONCEPT

Whilst recognising that an EUV spectrometer is an essential component of the Solar Orbiter, we must be aware that it must be compact and light-weight, must not be too telemetry 'thirsty' and must be able to cope with the thermal and particle environment of such an orbit.

A normal incidence system was originally envisaged for this spectrometer, to fit within reasonable length limits of the spacecraft. However, the thermal situation is extreme, in particular for a normal incidence design. Thus, two approaches are under consideration. The first is an off-axis normal incidence (NIS) system where a single paraboloid primary mirror reflects the selected portion of the solar image through a heat-stop to a spectrometer using a variable line spaced (VLS) grating in normal incidence. The second approach is a Wolter II design, again feeding light to a normal incidence VLS grating spectrometer, but using a flat mirror both to scan and to fold the instrument light-path into a reasonable length. Both designs are stigmatic.

The wavelength selections are geared to bright solar lines in the EUV from a broad range of temperatures.

The instrument structure would be made of carbon fibre, with silicon carbide (SiC) optical components. Multilayers will be considered if the final wavelength selection requires it.

Basic optical layout

In the off-axis NIS concept, the primary optical component is a paraboloid mirror of diameter 120 mm. The off-axis approach allows us to insert a heat-stop between the primary and the slit, which only allows a selected area of the solar image to the spectrometer. Most of the solar radiation is reflected by the heat-stop out of the front aperture. Thus, the thermal challenge for this design is almost exclusively concerned with the thermal control of the primary mirror itself.

The slit assembly lies at the focal plane, below the heat-stop, and beyond this is the spectrometer, with a toroidal variable line spacing (VLS) grating, forming a focus at a 2-D detector. There is no secondary mirror, as with a Ritchey-Chretien design, for example, and this helps to maintain a reasonable effective area. The VLS grating approach allows good off-axis performance compared to a uniform grating, and it brings the spectrometer 'arm' closer to the axis of the instrument, making the envelope smaller. The grating ruling spacing is yet to be decided but we have taken values up to 4800 l/mm as a guide for current design investigations. The use of a toroidal VLS grating allows a spectrometer magnification, with values of 2.5 being considered, whilst retaining optical quality.

Several wavelength bands are under consideration, namely 170-220 Å, 580-630 Å and >912 Å, and we anticipate obtaining up to two of these bands using either two orders or a split spectrometer (much like the CDS/SOHO approach). We have used the 580-630 Å band for the principal design discussions to date. Optimisation of the design is required, but this produces a basic envelope of 1.5 m x 0.4 m x 0.3 m.

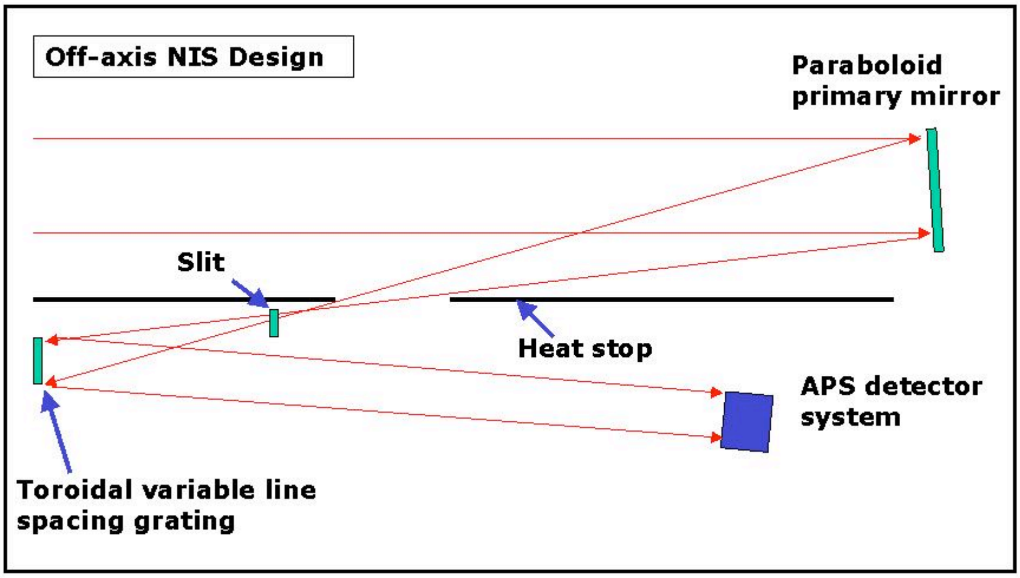


Figure 4.6: Optical scheme of the EUS: The Off-axis NIS Concept

The primary mirror presents a portion of the Sun at the slit, and it is this mirror that can be rotated to allow rastered images (i.e. exposures interlaced with mechanism movements to build up images simultaneously in selected wavelengths). Only a small fraction of the Sun will pass through the heat-stop to the slit assembly; possibly of order several hundredths of the disc area.

The instrument will not have independent pointing. It will be hard-mounted with the other remote sensing instruments, and used in conjunction with those instruments. Limited independent pointing can be performed using the primary mirror mechanism.

Calculations of the spot-size at the focus from ray-tracing codes have been performed for various angles of the primary, showing consistency with a 0.5 arcsec pixel size, which is baselined (equivalent to 0.1 arcsec at 1 AU when the spacecraft is at perihelion). This confirms that the concept of rastering with the primary is workable for the scientific requirements over the planned range. We plan a 5 μm detector pixel size and a field of view of order 34 x 34 arcmin.

The second design concept makes use of the paraboloid-hyperboloid reflections of a Wolter II telescope, i.e. at grazing incidence. Light is fed to a plane mirror before passing through a slit to a toroidal VLS grating and onto the 2-D detector The plane mirror reduces the length of the instrument but is also used to raster the solar image across the slit.

The advantage of the Wolter II approach is twofold. First, its grazing incidence optics provides better reflectivity and thus a better effective area than an equivalent NIS option. Second, the aperture can be considerably smaller making the thermal control for a mission such as Orbiter more manageable. However, the optical performance of the NIS approach is superior.

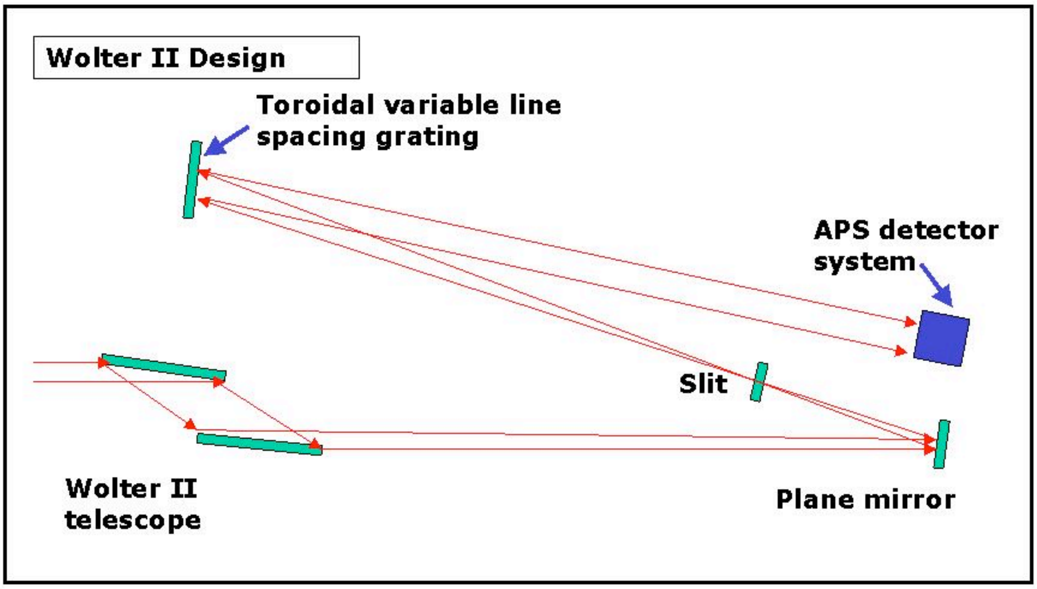


Figure 4.7: Optical scheme of the EUS: The Wolter II Design Concept

The heritage of these instrument concepts comes from the SOHO/CDS, SOHO/SUMER and Solar-B/EIS projects.

Both options would include a selection of slits, which can be chosen for particular observation programmes. In addition, it is assumed that the instrument resolution will demand an image stabilisation system which will involve motion of the primary mirror or plane mirror for the two concepts in response to an external signal.

Resolution/detector

We size the instrument to an observing distance of 0.2 AU (perihelion). A spatial resolving element of 0.1 arcsec represents 75 km on the Sun from the Earth; the same resolution can be achieved at 0.5 arcsec from 0.2 AU. With regard to spectral resolution, our basic aim is to return sufficient numbers of well separated spectral lines to allow thorough spectroscopic studies.

A detector array of 4k x 4k 5 m pixels is baselined. Thus, the EUS has a spectral range of 4 nm at 0.001 nm/pixel. The same array will give a spatial extent (vertical distance on the detector = slit length) of 0.5 arcsec x 4096 = 2048 arcsec = 34 arcmin. The solar diameter at 0.2 AU is 170 arcmin, i.e. we have a slit length of 0.2 of the solar diameter. For a given pointing location (spacecraft pointing), rastered imaging will be made up from movement in one direction of the primary mirror or the plane mirror, depending on the selected approach.

The choice of detector is dictated by the harsh particle environment which will be encountered by Solar Orbiter, as well as mass and power constraints.

The particle environment, which will be encountered by Orbiter, means that CCD-type detectors will most likely be inappropriate. We may anticipate a solar wind 'background' proton flux some 25x that of SOHO ($1/r^2$). For an average flux at 1 AU, of density 9 cm^{-3} (average speed and temperature of 300 km/s and 4 x 10^5 K (3.5 keV)) we expect 225 cm^{-3} at 0.2 AU. Thus the nominal particle environment will be similar to some modest storm events detected occasionally by SOHO.

There may also be an increased chance of encountering proton 'storms', due to vicinity, from shocks associated with mass ejection, with up to thousands of proton hits per second. One might expect events similar to those experienced by SOHO, with greater intensity, and, in addition, some near-Sun events may be generated by lateral expansion of CME disturbances (e.g. consider the EIT waves). The exact intensities remain unknown. The geometrical factors and magnetic configurations, which may play a role in defining the chance of occurrence of storms are also ill defined.

Also, we anticipate occasional impacts from solar flare neutrons whose 15.5 minute lifetime means that most missions do not encounter them. Finally, we anticipate a similar cosmic ray (non solar) flux to that at SOHO.

The net effect is an increase in particle hits, with some extreme conditions including occasional neutrons.

The radiation damage in CCDs is mostly caused by the creation of charge traps reducing the charge transfer efficiency (CTE). The radiation hardness of silicon Active Pixel Sensor (APS) detectors is much higher because CTE degradation is unimportant; charge is not transferred across the array using an APS detector, where on-chip electronics allows the extraction and amplification of charge from each pixel individually. The charge collection efficiency (CCE) may also degrade, but at higher radiation levels. Such a measurement should be performed once the type of detector has been chosen for any Solar Orbiter instrument.

The APS detector system is a realistic option for Solar Orbiter from a particle environment point of view. We note, however, that the on-chip electronics also provides additional low mass and power advantages, compared to CCDs.

The APS EUV sensitivity will be provided in the same way as with CCDs, in this case with back-thinned devices.

Mass

The instrument mass breakdown is given in the table. Given the fact that there are two design concepts still under discussion and that the optical designs and thermal designs need considerable optimisation, the mass estimates are necessarily preliminary. The breakdown assumes a secondary mirror, for this reason.

Unit	Mass (kg)
Primary mirror	0.5
Mirror support	0.3
Secondary mirror	0.1
Mirror scan mechanism	0.6
Slit assembly	0.3
Grating assembly	0.6
Image stabilization	1.5
Detector	1.0
Detector electronics	1.5
Baffles	0.5

Structure	5.4
Thermal subsystem	3.5
Harness	1.2
Margin	2.0
Electronics	6.0
Total	25.0

Table 4.4: EUS Mass breakdown

The total mass, at 25 kg, is light compared to similar instruments in operation. For example, CDS has a mass of 100 kg. However, CDS is a double spectrometer with an aluminium alloy structure, 5 individual detector systems and an independent pointing system. EUS will be a single spectrometer, with a smaller structure, made of carbon fibre, with SiC optics, one detector system and no independent pointing system. The EIS instrument under development for the Solar-B mission weighs in at 60 kg, but is 3 m long. Thus, a mass under 30 kg for a modern 1.5 m instrument appears to be quite possible.

4.2.2 Orbit, Operations and Pointing Requirements

The EUS instrument requires to be Sun pointed. It will be hard-mounted, along with the other remote –sensing instruments. The required co-alignment accuracy between instruments is 2 arcminutes, based on attaining a reasonable image overlap with the smallest instrument field of view. In addition, in operation, a pointing accuracy of 2 arcminutes is required. Fine pointing within the field of view of EUS can be achieved using the mirror mechanism, and stability will be achieved using a dedicated stabilisation system which uses a signal, probably from another instrument on Orbiter, to adjust the primary mirror or the plane mirror, depending on the design chosen.

Operations will be performed in pre-planned sequences stored, time-tagged in a deferred command store. The operations for an entire encounter (30 days) should be stored in this way. The sequences will have been selected during the period preceding the solar encounter. The planning and the selection of sequences will be done in concert with the other remote-sensing instruments. Indeed, target selection and pointing will be done as one instrument.

During the non-solar encounter periods of the orbit (the period outside the 30 days observation) the stored observational data will be telemetered to the ground. However, it would be highly desirable, if not essential, to practice encounter observations – even with considerably reduced telemetry – much as planetary encounter missions do, during the non-encounter parts of the orbit. In addition, any telemetry allocation, however low, for some modest synoptic observations during the rest of the orbit would be desirable.

4.2.3 Accommodation

The EUS instrument must be Sun-pointed and co-aligned relative to the other remote sensing instruments to an accuracy of 2 arcminutes. It must have an uninterrupted field of view of 34 arcminutes or greater; the aperture required is 120 mm in diameter. The thermal situation demands good views to deep space, making an ‘external’ mount with side and back views to space preferable. A radiator up to the size of the instrument footprint (1.5 m x 0.4 m) is envisaged. A separate electronics box must be mounted on the spacecraft. The main instrument would be mounted on kinematic legs and we envisage vents to outgas to space on a side or back panel.

4.2.4 Interface and Physical Resource Requirements

Telemetry

The nominal telemetry rate for the EUS instrument is 17 kbit/s. The full EUS detector image is 4kx4k pixels. At 12 bits per pixel, it will take 197 min to read one exposure. Since each exposure will form part of a raster, the raster cadence will be significantly longer. Studies from instruments such as CDS/SOHO have shown that careful line selection is far more important than data compression in managing the data return of such a spectrometer. Much of the spectrum is not required. Indeed, for specific studies, specific emission lines are required. A good rule of thumb from SOHO is that a selection of between 6 and 15 lines is good for most scientific purposes.

The EUS nominal resolving element is 0.5 arcsec along a 34 arcmin slit (4k pixels). The nominal spectral resolution is of order 0.01 Å/pixel. To obtain full line widths for million K lines, plus sufficient nearby background, one would want to return about 0.3 Å, i.e. 30 pixels.

The table shows a selection of potential cases. In each case a number of required lines is defined as is a selected length along the slit (spatial direction). The time to return such an exposure is given with a stated compression factor. The rastered image cadence is then given for four cases. The spatial length is given in pixels because of the varying distance to the Sun. We assume a return of 30 pixels across each line and 12 bit words.

No. of lines	Spatial Length	Compression Factor	Time to return exposure	Cadences for 200 (100 arcsec), 500 (250 arcsec), 1000 (500 arcsec) and 2000 (1000 arcsec) steps (minutes)			
6	500	10	6.35 sec	21	53	106	212
6	1000	10	12.7 sec	42	106	212	423
6	2000	10	25.4 sec	85	212	423	847
6	4000	10	50.8 sec	170	414	846	1694

This table shows that the telemetry rate is very restricting for the majority of rastered images. Given the dynamic/transient nature of the Sun's atmosphere, we should be looking for cadences of order minutes.

However, for some, specific solar applications, the figures of the above table are fine. This would include (i) small area (under 50x50 arcsec) rasters to investigate fine-structure dynamic events in the solar atmosphere; (ii) single-slit location sequences (i.e. no raster – the slit stays at one place monitoring intensities with one spatial dimension) to study transient intensity events such as blinkers and explosive events; and (iii) spectral atlas studies, which are seeking evidence for line identifications and discoveries using long-duration runs.

However, for many other studies, we do need to identify methods for improving the performance to avoid compromising the scientific return. Several options are possible, including:

- Increasing the telemetry rate: This option should be sought from the Project in any case;
- Returning line profile parameters rather than the full profile information;
- Returning image differences rather than full images.

These are just three options, one of which is beyond the control of the EUS study team! Returning the line profile parameters would reduce the data return by a factor of up to 0.1 (at best, 3 parameters instead of the 30 bins). Alternatively, we may wish to return 15 bins (2:1 summing) across the line rather than the full

30, if we are wary of the profile parameter method. The savings are illustrated below for the 1000 pixel length studies of the table above.

No. of lines	Spatial Length	Compression Factor	Time to return exposure	Cadences for 200 (100 arcsec), 500 (250 arcsec), 1000 (500 arcsec) and 2000 (1000 arcsec) steps (minutes)			
1. Basic Method							
6	1000	10	12.7 sec	42	106	212	423
2. 2:1 Line Profile Summing Method							
6	1000	10	6.3 sec	21	53	106	212
3. Line Parameter Method							
6	1000	10	1.3 sec	4	11	21	42

In addition to this, the feasibility of the image differencing method should be examined by any proposing EUS team. However, the figures in the table do now show cadences of order under 10 minutes and this is encouraging.

However, we note that the 17 kbit/s telemetry rate was a nominal value, allocated without detailed consideration. This figure is based on the 240 Gbit onboard memory for the payload and the single ground-station data dump scenario. In anticipation of improvements in ground-station coverage and on-board memory capacity, we briefly examine improvements in the telemetry rate of factors of 3 and 10, i.e. 51 kbit/s and 170 kbit/s.

No. of lines	Spatial Length	Compression Factor	Time to return exposure	Cadences for 200 (100 arcsec), 500 (250 arcsec), 1000 (500 arcsec) and 2000 (1000 arcsec) steps (minutes)			
Telemetry Rate of 51 kbit/s:							
6	4000	10	16.9 sec	56	140	280	560
6	1000	10	4.23 sec	14	35	70	140
Telemetry Rate of 170 kbit/s:							
6	4000	10	5.08 sec	17	70	140	280
6	1000	10	1.27 sec	4	17	35	70
Telemetry Rate of 51 kbit/s with Line Parameter Compresion Method:							
6	4000	10	1.6 sec	5.2	16	28	56
6	1000	10	0.4 sec	1.3	4	7	14

It is clear that there are very significant improvements in the performance with increased telemetry rates, providing reasonable rastered observations with cadences of under 10 minutes.

The figures demonstrate that the operation of an EUS instrument is feasible with the 17 kbit/s telemetry allocation given careful data selection and compression. The basic scientific goals of the instrument are not compromised but it is very restricting. It is clear that a greater telemetry allocation is highly desirable and would provide a MUCH improved scientific return. Thus, the ESA Project should consider possibilities for increasing the EUS telemetry rate allocation from 17 kbit/s; factors of 3 or 10 show significantly better scientific return, and it is felt that with increased onboard memory and improved ground coverage, this will be feasible. However, the EUS team must also study further methods for novel data selection and compression, such as returning image differences rather than raw images, and line profile parameters, to aid certain scientific study requirements.

Stability/pointing

Given our plan to achieve 0.5 arcsec resolution elements, we must choose one of the following options:

- do not include an image stabilisation system, assuming that the variations of the spacecraft stability occur on timescales much less than the exposure time of the spectrometer and thus any corrections could be done on the ground;
- include an image stabilisation system, possibly making use of VIM limb sensor error signals to drive adjustments to the secondary mirror.

For the present, the latter is assumed, with the central signal approach saving mass and complexity for the payload as a whole.

The EUS instrument would not require an independent pointing system; the remote sensing instruments are co-pointed and thus, hard-mounted together.

Thermal environment

At 1 AU the average solar intensity is 1.371 kW/m^2. During the nominal phase, in each 149 day period, the spacecraft will encounter a range from 2.142 kW/m^2 (0.8 AU) to 34.275 kW/m^2 (0.2 AU – 25 times the value at 1 AU). This presents a severe thermal challenge, which we tackle in a number of ways. The situation is much more extreme for the off-axis NIS approach, so we consider that concept here. A thermal study has been performed for the EUS (M. Gasquet, MSc thesis, Cranfield University/RAL, August 2002) and some important points are raised here.

The SiC primary mirror “sees” the full Sun. SiC optical components can run hotter than traditional components, but the primary receives about 390 W at 0.2 AU. Coatings can be used to reduce the absorption (to 0.2 for gold coating), though another option is to make the primary absorbing to reduce the thermal effects later in the optical system. Various options are under consideration. However, running the mirror at 61^{o}C, with a fixed radiator temperature of 50^{o}C, with a silicon-carbon mirror (absoptivity 0.1), the total radiator area required is 0.19 m^2 for the off-axis NIS design, i.e. 0.43 m x 0.43 m. However, this is a preliminary estimate and it does not include the radiator area for the APS detector. It does not include, also, any consideration of the thermal interface to the spacecraft. If the EUS radiator sees radiation from the back of the spacecraft heat-shield , the radiator size must increase. However, this estimate does suggest that the radiator size will be feasible.

For the Wolter II design, the aperture is much smaller. Using the same mirror materials, we find a requirement for a much smaller radiator, of order 0.012 m^2, for the primary mirror.

These estimates are for the situation at 0.2 AU. However, the thermal input varies considerably over the 150 day orbit – by a factor of 16. This can produce variations of over 100^{o}C over the orbit. To cope with this, a combination of heat-switches, heaters and heat-pipes are being considered. However, we note that the instrument will not run its prime scientific operation outside the encounter (30 day) periods, so it is not necessary to strive to attain perfect optical alignment during the aphelion periods, i.e. some flexing of the instrument is anticipated, with the best optical performance geared to the solar encounter periods.

Finally, we do note that the APS detector system should run cold and this would require a cold finger to a dedicated radiator.

4.2.5 Cleanliness, Ground Operations and Other Requirements

Cleanliness and Contamination

EUV optical surfaces in space are renowned for experiencing significant degradation in performance with time.

Under high irradiation, particularly in the ultraviolet, any contaminant deposited on an optical surface, even in very minute amounts, polymerizes, so the reflectivity of the surface drastically decreases. This effect is well known for synchrotron radiation optics as well as for some space instruments, but has been well avoided by the SOHO UV/EUV instruments. The degree to which the reflectivity decreases depends on the irradiation exposure and on the partial pressure of the contaminant.

However, for Solar Orbiter the situation is more difficult than it was for SOHO; due to the changing distance from the Sun, the level of UV irradiation will be higher and the thermal environment more variable.

Even with the most stringent procedures in the handling and assembling of the optical components, under the extreme irradiation conditions at 0.2 AU, there is a risk of a serious rapid degradation of the reflectivity, especially in the EUV. The variable thermal environment during the orbit makes evaporation and out-gassing from surfaces with increasing temperature unavoidable.

The decreasing reflectivity could be severe for optics at normal incidence, where the EUV reflectivity is relatively low and the EUV absorption is high. For example, gold could be a good candidate as an EUV coating for mirrors at normal incidence, since it has high visible reflectivity and also discrete EUV reflectivity (0.16 at 1200 Å and 0.13 at 600 Å), but a thin layer of contaminants deposited on its surface could drastically reduce the EUV response, and thus the effective area.

The effects could be less severe when the optics are used in grazing incidence. Firstly, the portion of the optics illuminated at grazing incidence is much larger than in normal incidence (for the same aperture) and correspondingly the flux decreases (this is beneficial also for cooling the optics); secondly, the effect of polymerization results in much less degradation of the reflectivity than in normal incidence.

It should be noted that the mirrors can be operated at relatively high temperature and this could help to reduce the deposition of contaminants.

It should be noted, also, that steps can be taken to reduce the levels of potential contamination, in space. The most important procedure would be a long out-gassing period prior to opening the instrument door. For the CDS and SUMER instruments on SOHO, the out-gassing period was 3 months from launch, and this was a deliberate (and successful) policy. With the inclusion of vents allowing out-gassing materials to escape, the long period certainly enabled the contamination to be reduced. Such a policy must be adopted for Solar Orbiter – possibly for several instruments. For efficient venting, the opening to space must be large (e.g. a partly opened aperture door, a door specifically designed for venting, or a permanent vent) and, in addition, the instrument interior must be preferentially heated (by passive or active heating).

Any EUV instrumentation must be developed with the most stringent contamination policy, both in the laboratory and in operation (e.g. out-gassing). Possible effects must be assessed thoroughly by the proposing teams and optical and procedural policies adopted.

Operations

The EUS instrument will have a deferred command facility and the ability to store command sequences. Thus, pre-planned observational sequences can be run, time-tagged, over long-periods. The planning for this can be done in an exercise similar to that adopted for CDS/SOHO. Individual observational routines will be designed using software tools and stored. One 30 day encounter may require the running of a series of such observations, with pointing and target selections decided with the rest of the remote sensing instruments.

We anticipate one principal planning meeting per orbit, to decide on the encounter strategy followed by detailed planning in accordance with that strategy. Shortly before the 30-day period starts, the sequences required should be uplinked and stored.

In addition, we anticipate a facility to make practice runs of the encounter observations on occasions during the non-encounter parts of the orbit. This need not require large telemetry allocation since it is designed to test the instrument operation rather than to return solar data.

4.2.6 Open Points and Critical Issues

There are four critical issues for the EUS, and these are issues which are in common with other instruments:

(i) Detector Development: The APS detector is baselined. To date there has been one 4kx4k 5 μm APS detector on the market, to our knowledge, but it is not appropriate for EUV or space use. The Detector Group at RAL in collaboration with E2V (Marconi, UK) have developed a 3kx5k 5μm APS device designed with back-thinning in mind. This has become available in December 2002. The RAL/E2V collaboration will develop the back-thinning process to ensure EUV sensitivity during 2003. To our knowledge, this is the only collaboration on the verge of producing back-thinned APS devices with the array sizes and pixel sizes required. The availability of a 4kx4k device by the time of the Solar Orbiter AO is highly likely. However, to ensure that the development includes a consideration of the needs of the Orbiter payload (not just EUS), it is recommended that ESA provide some support for this work.

(ii) Thermal Control/Design: The thermal design of the EUS is in early stages. Whilst a degree of feasibility is demonstrated, the design must be refined (including the input from spacecraft), and must cope with the variations of thermal input throughout the orbit in particular. This is being investigated.

(iii) Contamination/Degradation of Optical Surfaces: The harsh particle and thermal environment may have detrimental effects on optical surfaces, in particular multilayer coatings, other optical coatings and filters. Although much information has been acquired, it has been recommended that tests be made to understand the effects on specific surfaces (recommendation of the Solar Orbiter Payload Working Group).

(iv) Telemetry: The telemetry allocation is very restricting and requires very careful data selection and significant compression. Whilst the instrument can satisfy the scientific requirements, the increase in performance has been well demonstrated for any increase in the telemetry rate. The allocated telemetry rate was selected without detailed analysis, assuming a 240 Gbit memory for the spacecraft and one ground-station. Any increase in storage, ground-station coverage, and, thus, the allocated telemetry would be a very significant scientific advance.

Name / acronym	**Extreme Ultraviolet Spectrometer / EUS**

Objectives	1) to provide plasma diagnostic observations of solar plasmas over a broad temperature range from chromosphere to corona for the study of all solar atmospheric phenomena, using EUV spectroscopy
	2) to identify fundamental processes at work on all scales in the solar atmosphere

General description	Grazing incidence or normal incidence telescope system feeding a variable line spaced grating spectrometer incorporating active pixel sensor detectors.

Reference P/L and/or heritage	New generation of EUV spectrometer building on the successful CDS/SOHO, SUMER/SOHO and EIS/Solar-B instruments, with the same teams.
	Potential consortium has long heritage of EUV and X-ray solar spectroscopic instrumentation, including SMM (1980-89), CHASE (1985), Yohkoh (1991-2001), SOHO (1995+), SERTS, Solar-B (2005), SDO
	Potential hardware contributions for key technology groups with heritage in key areas (e.g. APS detectors from RAL detector group, gratings from Padua University, SiC optical components from Max Planck, Lindau)

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	1-2	Active Pixel Sensor (APS)	4kx4k array, 5micron pixels, back thinned for EUV sensitivity
Spectral range	Å	2-3 bands in range 170-1000	Possibly 1 range with 2 orders.
Operating T	K	- 80 degrees	
Optics / antennas			
Type	N/A	1-2 mirror reflections plus grating	SiC
FOV	arcmin	Up to 34	
Bandpass	Å	170-1000	2-3 bands in range; EUV selected by filter or microchannel plate
Pointing	N/A	Sun directed	Directed at solar targets with rest of remote sensing payload
Configuration			
Physical Units	No	1 plus electronics box	
Unit layout	N/A	Adjacent	
S/C related requirements	N/A	Aperture required (120 mm) in front panel; radiator area up to size of instrument footprint on side facing panel; leg mounts; vent to space on side or back panel.	
Physical			
Mass, total	kg	25	
Mass unit 1	kg	19	Carbon fiber structure
Mass unit 2	kg	6	
Dimension 1	cm	160x30x40	
Dimension 2	cm	20x20x20	
Power			
Average	W	25	
Peak power	W	TBD	
Stand-by	W	TBD	
Data rate / volume			
Average data rate	kbit/sec	17 (Pre-assessment study allocation)	Note: 17 kbit/s was the allocated value in the Pre-Assessment study. This is

		>51 (highly desirable)	very restricting. See note above.
Peak data rate	Bits/sec	TBD	
Data volume /orbit	GByte	5.5 [16.5]	Lower figure assumes 17 kbit/s over 30 days. Bracketed figure assumes 51 kbit/s
Own data storage	MByte	TBD	
Thermal			
Heat load to radiator	W	170	
Operating T range	K	Radiator >50°C Optics <100°C Detectors –80°C	Cool detectors, warm optics
Other requirements	N/A		
Cleanliness			
EMC requirements	N/A	Not stringent	Normal s/c equipment levels
DC magnetic	N/A	Not stringent	Normal s/c equipment levels
Particulate	N/A	BOL 85 ppm EOL 150 ppm	SOHO levels required
Molecular	N/A	BOL 5 x 10^{-8} gms/cm^2 EOL 1 x 10^{-7} gms/cm^2	SOHO levels required
Miscellaneous			
Mechanisms	No.	1 door; 1 scan mirror mech.; 1 image stabilisation mirror mech.; 1 slit change mechanism;	
Alignment	Arcmin	Absolute pointing to 2 arcmin; Co-alignment relative to other instruments 2 arcmin.	
Unobstructed FOV	deg	34 arcmin	
Orbit requirements		None	
AIT/AIV requirements		TBD	

Development approach / schedule

Preferred model philosophy	EM, QM, FM
Estimated development time	EM (1 yr), QM (2 yr), FM (1 yr).

Critical areas: Technology readiness – Design maturity level

Critical area /unit/ subsystem	TRL	DML	Justification and remarks
Detectors	4	4	CCD detectors not appropriate for environment. APS detectors selected to cope with environment and to provide required small pixels. Appropriate 4kx4k devices under development but not available yet. We are associated with the development of these devices, which must be supported.
Thermal control	2	5	Solar Orbiter's extreme thermal environment and variations have demanded some modelling which will be followed by further modelling and test activities.
Contamination/Degradation	2	5	Degradation of optical surfaces (multilayers, coatings and filters) is of concern under such extreme thermal and particle environments. Requires some test activities.

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in

relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

4.3 EUV Imager (EUI)

4.3.1 Instrument Description

SCIENTIFIC DRIVERS

Observations from Yohkoh, SOHO and TRACE in the extreme ultraviolet and soft X-ray wavelengths have revealed a truly complex, highly dynamic solar atmosphere with magnetic loops confining plasmas at widely varying temperatures. The TRACE EUV observations in particular illustrate the existence of fine-scale structures in coronal loops, and reveal continuous dynamic activity at the smallest scales. In the quiet Sun, various "events" of different sizes (e.g., bright points, explosive events, jets, blinkers) all provide evidence for small-scale heating and morphological reorganization, probably related to magnetic reconnection. The observed distribution functions have self-similarity properties, which point at sub-resolution processes. The results from Yohkoh, SOHO, and TRACE led to new questions concerning the real basic dimensions of coronal structures, the role played by nanoflares in the heating of the quiet solar corona, and the structuring of the corona above the poles.

The observations from Yohkoh, SOHO and TRACE have also taught us that these small scale structures and processes have far-reaching, global consequences, effecting large scale structures thousands of Mm away, on the other side of the Sun. With Solar Orbiter, dramatically improved understanding can be obtained by simultaneously increasing the spatial resolution by an order of magnitude, while maintaining the full Sun global context in order to connect the observed chromospheric/coronal phenomena with close *in-situ* measurements.

This is where an EUV Imager (EUI) on the Solar Orbiter can fully exploit the unique capabilities of the mission. EUI will provide EUV images with an order of magnitude higher spatial resolution than TRACE in order to reveal the fine-scale structure of coronal features. The closeness of the Sun makes this task easier. EUI will also provide full disc EUV images of the Sun in order to reveal the global structure and irradiance of inaccessible regions such as the "far side" of the Sun and the polar regions. It will also be the invaluable tool for pioneering the connection between in situ and remote sensing observations.

The European solar physics community has a well-established expertise in solar EUV/UV imaging as illustrated by the successful EIT instrument (Delaboudiniere et al. 1995)on SOHO, as well as the EUV Imager for SDO (SHARPP, Defise et al. 2002). Additional experience has been gained in the areas of the thermal (radiative) load on the optics, as well as instrument miniaturization as a result of design studies for the integrated, remote sensing package on Solar Probe (Hassler et al. 2000).

INSTRUMENT CONCEPT

A single telescope, providing both high spatial resolution and a full disc field-of-view, faces two difficulties:

1. optical design (PSF/FOV), limited number of pixels on current detectors and telemetry constraints;
2. thermal (radiative) load at perihelion (25 solar constants on the first optical element if the field-of-view is not reduced).

These considerations led to a separation of the EUI into two instruments: a High Resolution Imager (HRI) and a Full Sun Imager (FSI). In this way, one can find specific solutions for the thermal load and have realistic optics and detectors.

High Resolution Imager (HRI)

The design driver is the need to get rid of the unwanted solar radiation, which goes along the direct light path. A first step consists in reducing the entrance aperture of the instrument to about 2 cm and to fully use the available length of the payload module. In doing so, one can cluster a few telescopes together, each one dedicated to one wavelength, which eliminates any moving part, takes full use of the aperture and allows for *simultaneous* observations in different wavelengths.

Essentially one reduces the FOV through the use of an entrance baffle and one sets the FOV to 500 arcsec (70 000 km at perihelion) for a 2 x 2 K detector. This sets the equivalent pixel size to 0.25 arcsec (equivalent to 0.05 arcsec from Earth, 35 km on the Sun at perihelion) [should be left somewhat open at A.O level?]. The pixel size of the detector is baselined at 9 □, so the equivalent focal length is 7.2 m.

We baseline a set of three telescopes working at 13.3, 17.4, and 30.4 nm. These three wavelengths cover a very wide range of temperatures (from 5 x 10^4 K up to 1.6 x 10^7 K) and targets (from quiet Sun to flares). Note that the 13.3 nm band includes a very hot line (Fe XXIII), visible only during flares and that the 17.4 nm band only selects the 17.4 nm Fe X line, excluding the cooler Fe IX line at 17.1 nm. [should be left somewhat open at A.O level?]

The basic feature of each telescope relies upon the use of a long baffle to reduce the FOV and the stray light, an entrance metallic foil filter and an off-axis Gregory telescope with multilayers fitted to each wavelength. The Gregory concept allows to put stops at the focal plane and at the image of the entrance aperture, which leads to a significant reduction of straylight. Figure 4.8 shows the optical scheme of the HRI.

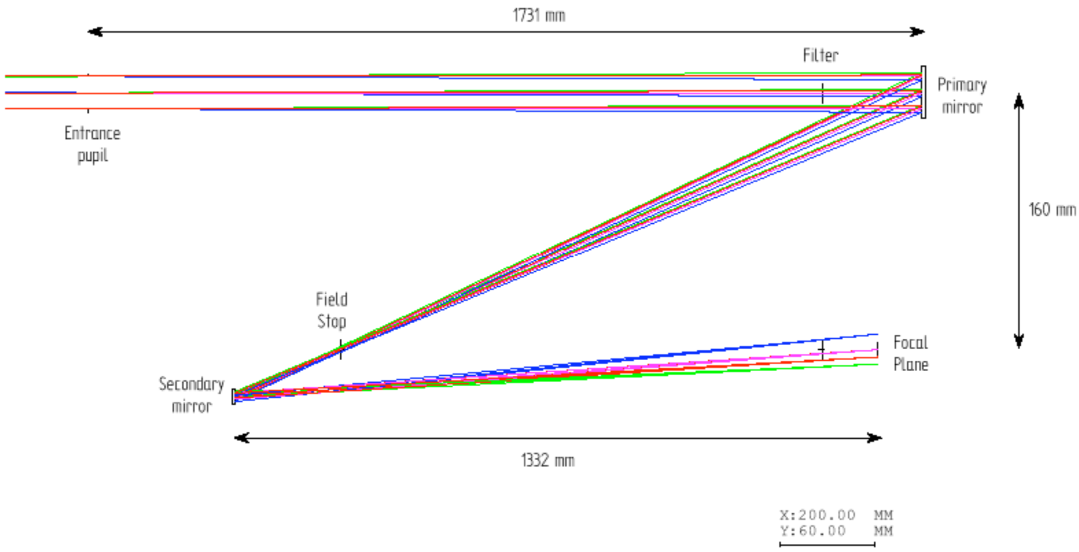


Figure 4.8:HRI optical scheme with its basic components.

A carbon-carbon composite structure is planned.

We baseline 2k x 2k pixel APS detectors (see EUS section). Diamond or GaN/AlGaN new generation detectors are interesting alternatives. They are robust, solar-blind (allowing a better instrumental sensitivity) and rad-hard.

For the 17.4 nm and 30.4 nm channels, the small aperture will give a few counts per pixel per second. [This is quite a serious issue. If we bin, we are just giving away the resolution we were after! As for augmenting exposure time, we enter the sub-sampling mode that you have seen me describing with graphs (10Hz required at 35km pixel). It's perhaps better than nothing but we'll then miss the dynamics and possibly get kinematic blurring, ie. not more SNR. I don't say it is a real show-stopper but close… It's at least a crucial driver for HRI, and as PWG, we should attract ESA's attention and call for further study of that issue. I personally think there are ways to cope with the shortage of SNR, but not that straightforward. Or shall we leave this open to proposing teams?] The count number for the 13.3 nm channel can be much higher but only on very active regions and flares.

The electronics (including the DPU) is common to the three detectors and to the FSI telescope. Its volume should fit within the overall volume given in the Table 4.5.

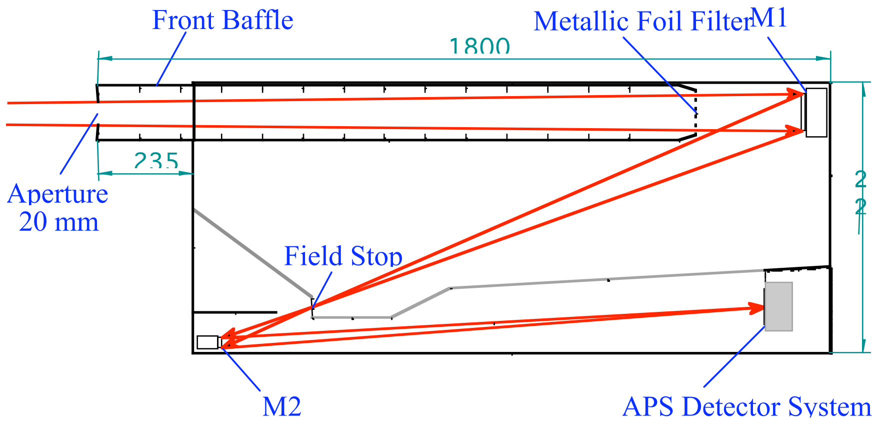


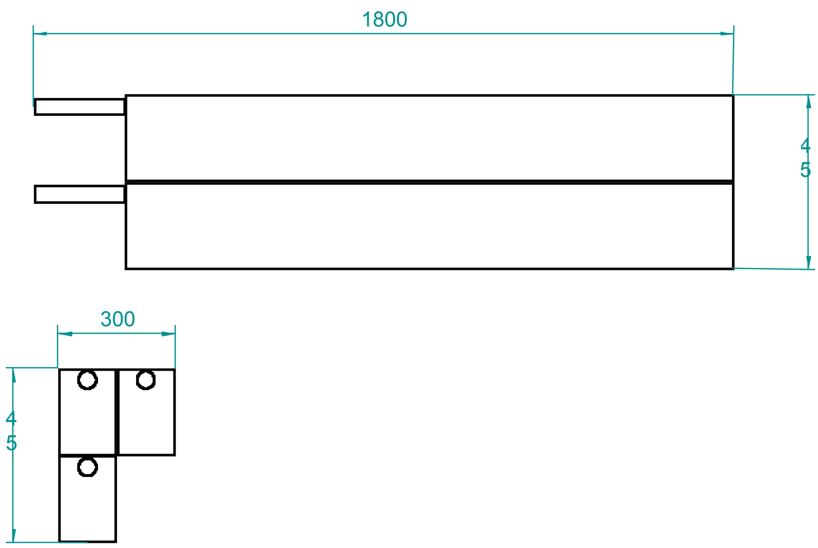
Figure 4.9: HRI overall accommodation with its basic components. (vertical scaling is 3 X horizontal scaling)

Figure 4.10: HRI possible implementation of the three telescopes

Full Sun Imager (FSI)

The Full Sun Imager needs a wide field (about 5°) when at perihelion in order to cover the full lower solar corona with about 50 % margin in each direction. [It could be made 4° (=3 Arctan(700/(150000/5)) if there was an asymmetrical design taking advantage of a preferred direction for offpoints, If not we really need 5.4° FOV to still always see the full disk when at the limb!] An off-axis Gregorian optical system has been selected to reduce the field curvature aberration in the large 5° FOV. This provides an RMS spot diameter smaller than 9 µm, compatible with the detector pixel size. Figure 4.11 shows the optical scheme of the FSI.

Similarly to HRI, a front baffle is used to protect the metal foil filter from full sun heat load. The current design can provide a limitation from 3.3 (at 1190 mm from entrance of the baffle) to 1.7 (near M1) solar



constants, depending on the location of the filter, as shown in Figure 4.12.

Other innovative designs based on Fresnel zone plates, photon sieve cameras, ... are currently under evaluation, and might replace this 1.8 m length instrument.

Figure 4.11: FSI optical scheme

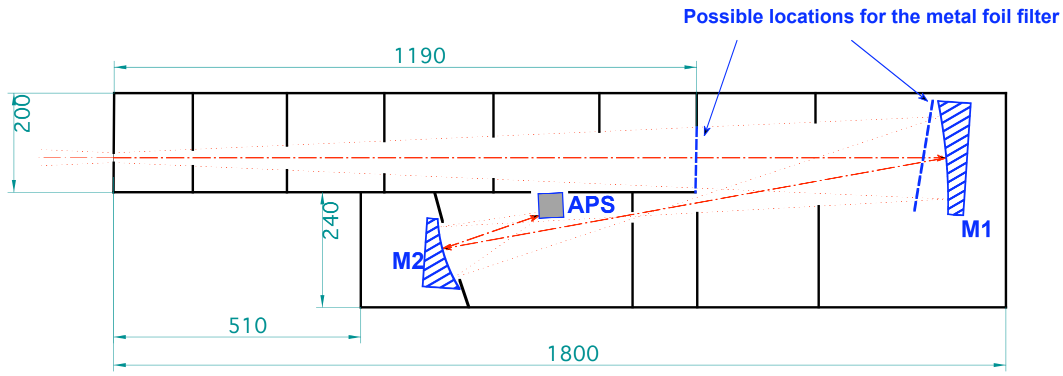


Figure 4.12: FSI instrument, with its components, the 2 mirrors M1 and M2 and the APS detector system

4.3.2 Orbit, Operations and Pointing Requirements

HRI stability/pointing

Given our plan to achieve 0.25 arcsec resolution elements and the fact that exposure times are of the order of 10 seconds, an image stabilisation system should be included, possibly making use of the VIM limb sensor error signals to drive adjustments to the secondary mirror.

The three telescope cluster has its own pointing mechanism since the limitation of the FOV prevents any independent internal pointing. [This is a bit confusing in the context of a globally pointing payload] The pointing is performed by a joint displacement of the three telescopes. The spacecraft has an offset capability of about 10°, much larger than that required (2.5° at perihelion), and thus in principle it would be possible to point to a solar target using the whole spacecraft. [I understand this was "decided" at the first PWG mtg.]

4.3.3 Accommodation

4.3.4 Interface and Physical Resource Requirements

HRI telemetry

With all three telescopes working together at a 10 seconds cadence, we arrive at a *peak* telemetry rate of 720 kb/s after data compression. An average rate of less than 20 kb/s can be obtained by a combination of further compression/selection schemes (e.g., or cascade time differentiation) and by adopting appropriate observing strategies (e.g. interleaved high/low time resolution sequences, observations in only one channel, etc.). We assumed above that we can compress the data by a factor 20. Actually, with a Rosetta-type compression scheme, tests performed on solar images indicate that compression factors as high as 50 can be achieved.

The concept of peak TM rate is not so relevant since we are often out of contact when HRI is operating. In Utopia, I would baseline an *average* cadence of 10Hz, and hence 3x2k^2x14bitx10Hz/20=84 Mbps (FSI is suppose to contribute negligibly to the TM budget). We then still lack a factor >4000. I see the solution coming from: better lossy compression scheme (a factor say 4?), and YES, "a priori" AND "a posteriori" selection schemes. We need to have intelligent pointing (from FSI images (+HMI?)) 1/to point at areas bright enough, if there is a shortage of signal, 2/to increase our chances to catch all kinds of features without relying on mere luck, 3/to be able to find and observe the source of the in situ observables (could be made a mission goal!). "A posteriori" selection is important too, in order to get the pre-event records, and to optimise TM. Note that such a selection scheme increases the detectors' duty cycle in the UV. Once we have image processing onboard abilities, the "a posteriori" is granted, while the "a priori" does raise mission security issues. However, the S/C will have to manage autonomously when at the backside, is it that more difficult to point a BP than the centre of the disk? More in the Harra/Hochedez and Harrison/Hochedez AI reports. Finally additional telemetry (and on-board memory) would help a lot. This must be further requested and assessed

HRI mass budget

The overall HRI mass budget is given in the Table 4.5.

Component	**Mass (kg)**
Structure (including the baffle) :	4.8
Optics (including mounts) :	0.7
Detector (including electronics and shielding) :	0.5
MLI and radiator	2.2
Margin (20 %)	1.5
Total per telescope	9.2
Total for 3 telescopes	29.1
Electronics shared between the 4 telescopes	3.0
Grand total for the HRI	32.1

Table 4.5: Mass breakdown of the HRI of EUI

HRI power

The power requirements have been considered for the detector, electronics and the pointing actuator (common to the three telescopes). For a given telescope, we anticipate a constant power requirement of about 4 W for the cycling (power on, integration, dump to memory). With reasonable margins and the required power for pointing, the total is 5 W. If the three telescopes are working simultaneously, they will consume about 15 W for basic functions (above) to which one has to add about 2 W for data compression and dumping by the common DPU. Non-simultaneous (sequential) observations of course lead to a smaller power consumption. [More DPU power will mean more electrical power. I do not have the MIP<->Watt conversion factor. Could 100MIPs be a starting point?]

FSI requirements

An average rate of 0.5 kb/s is sufficient to transmit one compressed full Sun image every 4800 s (about every hour and 20 minutes).

The FSI mass is about 10.5 kg (structure: 6.5 kg; optics and mountings: 3 kg; detector and associated electronics: 1 kg).

Power is only required for the detector (3 W) since the electronics is common with HRI.

The FSI also relies on the pointing stability of the payload module but with constraints much less stringent than for the HRI.

The HRI and FSI characteristics are summarised in the Tables 4.6 and 4.7, respectively..

3 Telescopes	Off-axis Gregory; 20 mm diam.
Spatial resolving element	0.25 arcsec pixel size (35 km on Sun at perihelion)
Pointing	Common pointing mechanism for the three components
Detector	Baseline 9□, 2000 x 2000 pixels
Field of View	70x70 Mm^2 at perihelion
Peak Telemetry	720 kb/s (with compression 20) irrelevant? Why should we ever transmit one image per exposure time?
Average Telemetry	19.5 kb/s
Mass	30.1 kg
Power	17 W (essentially detectors; shared electronics and DPU)
Size	Instrument = 3 X (1800mm x 150 mm x 150 mm)
Thermal	Operating temperature 20°C with passive cooling; 140 cm long baffle at the entrance of each telescope
Stability	Active image stabilisation at secondary.

Table 4.6: Characteristics of the HRI of EUI

Telescope	Off-axis Gregory; 20 mm diam.
Spatial resolving element	9 arcsec (1300 km on Sun at perihelion)
Pointing	Pointed at Sun centre
Detector	Baseline - 9 □, 2000 x 2000 detector
Field of View	4 x 4 solar radii at perihelion
Peak Telemetry	240 kb/s (with compression 20)
Average Telemetry	0.5 kb/s
Mass	10.5 kg
Power	3 W
Size	Instrument = 1800 mm x 440 mm x 250 mm
Thermal	Operating temperature 20°C with passive cooling

Table 4.7: Characteristics of the FSI of EUI

INSTRUMENT SUMMARY

HRI and FSI	Off-axis Gregory Telescopes; 20 mm diam.

Spatial resolving element	HRI : 0.25 arcsec pixel size (35 km on Sun at perihelion)
	FSI : 9 arcsec (1300 km on Sun at perihelion)
Pointing	HRI : pointing mechanism to point anywhere on the Sun
	FSI : pointed at Sun centre
Detector	Baseline - 9 □, 2000 x 2000 pixels
Field of View	HRI : 70x70 Mm^2 at perihelion
	FSI : 4 x 4 solar radii at perihelion
Average Telemetry	20 kb/s
Mass	40.6 kg
Power	20 W
Size	see previous tables
Thermal	Operating temperature 20°C with passive cooling

Table 4.8: EUI instrument summary.

4.3.5 Cleanliness, Ground Operations and Other Requirements

4.3.6 Open Points and Critical Issues

- Imaging detectors, their EUV sensitivity, stability, solar-blindness, radiation hardness, format, pixel size, readout speed, etc.
- Telemetry and memory allocation
- Image processing: exact scope (pointing, post-selection), robustness, CPU & memory requirements
- Photon sieve design, and any design bringing more photons (per kg) to the focal plane.
- Observing strategy: FSI should be observing all around the orbit in a somewhat synoptic way because the higher latitudes and the far solar side do not occur necessarily at perihelion, and also for EPO reasons.
- Ageing of the metal foil filters under 2 solar constants.
- Mechanical behavior of the foil filter under vibrations and venting launch environment, when mounted at the extremity of a 1.5 m length cavity.

4.3.6.1 A list of areas of concern and future investigations and technologies are listed below for the thermal design:

(1) Optimisation of the optical designs

Both telescope designs will fit inside the current available payload envelope. The length of the HRI is critical in the sense that it only just fits within the payload bay. For more confidence we need to study more detailed geometry of optics and electronics boxes. The overall optical design of the telescope can perhaps be optimised a bit more if needed.

(2) Investigation into the stressing of lightweight materials.

The design of the telescopes needs to be extremely lightweight; we need to look in more detail into the stressing of the proposed telescope designs. This would involve an investigation into the different lightweight materials that could be used.

(3) Thermal modelling with the spacecraft

The first analysis data show some interesting results. The stiffness of the structure is in the right ballpark. The thermal model shows that the structure is following the temperature of the spacecraft. We need the results of the (coupled) s/c analysis to see if this is realistic.

(4) Temperature of the entrance filters.

A potential significant problem is the (first) filter. This holds for both telescopes, the filter temperature can rise to temperatures well above 450 °C. This may not be acceptable we need to look into possible filter designs and coatings. The location of the first filter is critical, by far the best location of the first filter is at the entry aperture of the telescope. If that is acceptable we can lose the baffle for the FSI, which will save mass.

(5) *Alignment of optical components during large temperature variations*.

If the temperature of the s/c (interface) changes significantly during the orbit, which is likely, the temperature of the structure will follow that variation. As shown for the FSI a 10 °C swing will generate a 4 micro meter relative displacement between optical components. For HRI this will be twice is high. If this is unavoidable we may need to use mechanisms to compensate these misalignments.

(6) The use of smart/adaptive optics to correct for optical misalignment.

It would be prudent to investigate the use of active components to adjust the alignment of optical elements in orbit. For example, deformable mirrors can be lighter than monolithic mirrors and can adjust the focus of the optical system. Various low-precision actuators have been used in space already, but high precision actuators would be need for Solar Orbiter, and more study needs done on this. The power requirements of different adaptive systems need to be investigated.

(7) Investigation of the temperature of optical components.

The results so far are encouraging but more detailed analyses and also more detailed coupled analyses are needed to size problems more accurately. For the thermal model a more refined model is needed to investigate the temperatures of the individual components.

(8) Investigation into materials used in the MLI

The MLI must be able to withstand the temperatures reached.

(9) Investigation into thermochromatic coatings for radiators

A thermochromatic surface on a radiator would allow its emittance to vary during the orbit, with no moving parts or control systems necessary. This would adjust for large variations in temperatures, if this proves to be a problem for the optical components.

(10) FSI thermal study

The major object of the modelling was to provide a representation of the FSI in the Solar Orbiter thermal model. This has been achieved, with three alternative models. It should be noted that if any internal data is to be gleaned from these models, they need to be reviewed in detail for the areas in question.

It is fairly clear that the baffle design considered in the thermal study did not appear to deal well with the high incident solar radiation, and some more thought needs to be given to this aspect of the design. The second and third models illustrate two methods of improving the situation.

It is also suggested that the performance of the thermal filter could be enhanced by changing its solar facing surface from a first surface mirror to a second surface mirror. This is demonstated in a fourth model. This could be achieved by coating this surface with a few microns of polymer such as Teflon. The coating thickness is critical, but an absorptivity/emissivity ratio of 0.3/0.8 should be achievable by this means, which would dramatically reduce the thermal filter temperature. The production of such a component should be carried out by a specialist, and could be very expensive. The problem of atomic oxygen erosion would also have to be investigated for the proposed orbital conditions.

(11) FSI Mass Budget

The mass budget of FSI needs to be verified and may require some updates.

Name / acronym	Extreme Ultraviolet Imager / EUI

Objectives	1) to reveal the fine scale structure and dynamics of solar plasmas over a broad temperature range from the chromosphere to the corona.
	2) to identify fundamental processes at work on all scales in the solar atmosphere.
	3) to link with in situ observations

General description	High Resolution Imager (HRI): 3 off-axis Gregory filtergraph telescopes Ful Sun Imager (FSI): 1 off-axis Gregory filtergraph telescope The 4 sensors incorporate large format active pixel sensor (diamond?) detectors.

Reference P/L and/or heritage	SOHO/EIT, TRACE, SDO/AIA

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type		backside thinned Diamond (?) or Si Active Pixel Sensor (APS)	4kx4k format array
Spectral range	Å	4 EUV bands	
Operating T	C	- 80 deg (Si APS) -10 deg. (Diamond APS)	
Optics / antennas			
Type	N/A	2 mirror reflections per channel	
FOV	arcmin	320 x 320 (FSI), 6 x 6 (HRI)	(all parts of image are vignetted with known vignetting function produced by front baffle)
Spectral range	Å	HRI: 133, 174 & 304 Å FSI: TBD between 171 to 304 Å	
Bandpass	Å	10-30 angstroms per channel	3-4 channels; selected by multilayer coatings and filters
Pointing	N/A	Sun directed, co-aligned with S/C	co-aligned with other remote sensing instruments
Configuration			
Physical Units		4 telescopes + 1 electronics box	
Unit layout	N/A	Adjacent	
S/C related requirements	N/A	+ co-aligned with S/C and other instruments + cover mechanism at the entrance of each baffle	
Physical			
Mass, total	kg	48.6	
Mass unit 1	kg	40.6 (FSI+HRI)	
Mass unit 2	kg	8 (electronics box)	
Dimension 1	cm	3 X (180 x 15 x 15) 1 X (180 x 44 x 25)	this volume includes the 4 units of FSI and HRI, with the optical baffles
Dimension 2	cm	20x20x20 (electronics box)	
Power			
Average	W	25	
Peak power	W	TBD	
Stand-by	W	TBD	
Data rate / volume			
Average data rate	kbit/sec	20	

Peak data rate	Bits/sec	TBD	
Data volume /orbit	KByte	TBD	
Own data storage	MByte	TBD	
Thermal			
Heat load to radiator	W	170	
Operating T range	K	Radiator >50°C Optics <100°C Detectors –80°C (Si APS) or -10°C (Diamond APS)	1.5 m baffle in front of primary mirrors to limit thermal load on foil filters to ~ 2 solar constants.
Other requirements	N/A		
Cleanliness			
EMC requirements	N/A	Not stringent	Normal s/c equipment levels
DC magnetic	N/A	Not stringent	Normal s/c equipment levels
Particulate	N/A	BOL 85 ppm EOL 150 ppm	SOHO levels required
Molecular	N/A	BOL 5×10^{-8} gms/cm^2 EOL 1×10^{-7} gms/cm^2	SOHO levels required
Miscellaneous			
Mechanisms	No.	HRI: 1 door; 1 image stabilisation mirror mech per channel FSI: 1 door	
Alignment	Arcmin	Absolute pointing to 1 arcmin; Pointing relative to other instruments 20 arcsec.	
Unobstructed FOV	deg	4.5	
Orbit requirements		None	
AIT/AIV requirements		TBD	

Development approach / schedule

Preferred model philosophy	EM, QM, FM
Estimated development time	EM (1 yr), QM (2 yr), FM (1 yr).

Critical areas: Technology readiness – Design maturity level

Critical area /unit/ subsystem	TRL	DML	Justification and remarks
Detectors	2	4	Diamond detectors are baselined to operate at +20 deg. C, thus removing the -80 deg. C cooling requirement. However, these devices require the greatest technology development. Otherwise, 4K x 4K back-side thinned Si APS detectors will be used for radiation tolerance and to provide pixel by pixel variable gain adjustment. Note that a 4k x 4k format APS is still an ambitious technology development item for space research applications. Large format back-side thinned CCD cameras will be used as backup, with increased resource requirements.
Thermal design	5	3	There needs to be detailed modelling of the instruments with the spacecraft design to allow a determination of the impact of the orbital fluctuations on, for example, the stability of the optical layout and stressing on materials, filters.
Filters (at 2 solar constants) (at 8 solar constants) (at 20 solar constants)	10 8 6	3 3 3	There is a potential problem with the temperature of the first filter, and initial modelling show that the temperatures of the filter can rise to greater than 450C. The materials and coatings should be investigated. If a filter can withstand a higher temperature then for the

			FSI at least, the size of the instrument can be reduced.
Smart optics	6	3	Active components can be used to adjust the alignment of optical elements in orbit. Only low precision actuators have been used in space to date, but high precision actuators would need to be used for solar orbiter. In addition the power requirements need to be investigated. Using such optics would reduce the mass of the system.
Thermochromatic coatings	3	3	Such coatings could be used on a radiator to allow the emittance to vary during the orbit, with no moving parts or control systems. This could provide a means of adjusting for large temperature variations during the orbit.
Compression techniques	5	3	The limitations of this mission require high compression techniques to reach compressions of greater than 20. These techniques have to be effective when the data is noisey due to radiation etc.
Image recognition	4	3	It may prove necessary to have autonomous pointing control of the spacecraft when, for example, the spacecraft is at the far side of the Sun. This would be used for hunting a bright active region etc. Basic autonomous flare triggers have been used for instrument pointing in previous missions and are being developed for future missions (e.g. solar B and SDO). Any procedure used would have to be 100% safe, to avoid any risk to the spacecraft.

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

4.4 Ultraviolet and Visible-light Coronagraph (UVC)

4.4.1 Instrument Description

SCIENTIFIC DRIVERS

Co-rotation, during the helio-synchronous phases of the orbit, will freeze for many days coronal structures in the plane of the sky. This condition will be most favourable to investigate the evolution of the magnetic configuration of streamers in order to test the hypothesis of magnetic reconnection as one of the main processes leading to the formation of the slow solar wind.

The out-of-ecliptic vantage point will allow for the first time a unique view of the plasma distribution and solar wind expansion in the coronal low-latitude/equatorial belt. Therefore it will be possible to measure the longitudinal extent of coronal streamers and coronal mass ejections. These parameters, that at present are unknown, are essential to determining the magnetic flux carried by plasmoids and coronal mass ejections in the heliosphere.

UVC will allow the first determination of the absolute abundance (i.e. relative to hydrogen) of helium in corona. Helium, the second largest contributor to the density of coronal plasma, is important for the dynamics of solar wind, and it may act as a regulator to maintain a nearly constant solar wind mass flux.

UVC will also determine the differential outflow speed of the major components (H, He) of the solar wind and discriminate the mechanisms of solar wind acceleration.

The European solar community has a well established expertise in visible light and UV coronagraphy as illustrated by the successful LASCO and UVCS instruments on SOHO.

INSTRUMENT CONCEPT

UVC is an externally occulted telescope designed for narrow-band imaging of the EUV corona in the He II 30.4 nm and H I 121.6 nm lines, and for broad-band polarisation imaging of the visible K-corona, in an annular field of view between 1.2 and 3.5 solar radii, when the Solar Orbiter perihelion is 0.21 AU. When it moves to 0.3 AU, at the end of the nominal mission, the UVC field-of-view is within 1.8 and 5.3 solar radii.

The telescope optical configuration is an off-axis Gregorian. The EUV L□ lines are separated with multi-layer mirror coatings and EUV transmission filters. These mirrors with coating optimised for 30.4 nm still have a good reflectivity at 1216 nm and in the visible. The visible light channel includes an achromatic polarimeter, based on electro-optically modulated liquid crystals.

Optical design

-Telescope

The external occulter ensures both thermal protection for the optics and better stray-light rejection. A Gregorian telescope design has been chosen because it gives real images of the external occulter and the edges of the telescope primary mirror. Field and Lyot stops will be used to reduce stray light and diffracted radiation from the telescope primary mirror. In addition, use of an annular secondary mirror and light trap captures and removes diffracted light from the external occulting disc and its supports. A light trap behind the primary mirror ensures that only the light reflected by the mirror enters the filters and detector assembly structure. The direct light from the solar disc is mostly dumped into space, though a portion of it may be used for radiometry and / or coarse imaging of the solar disc in the three wavelength bandpasses.

-UV and Visible-Light Channels

The instrument structure is made of carbon fibre, with Zerodur optical components. Zerodur is used for the mirrors' substrate because of its extremely small thermal expansion coefficient. Optics made with this material is also easy to polish into non-spherical figures with sub-angstrom rms surface roughness.

The basic specifications and drivers of the instrument are summarised in Table 4.10.

Structure

The instrument structure is of carbon fibre/cyanate or carbon-carbon composite, designed for zero coefficient of thermal expansion in the axial direction. It is in the form of a rigid rectangular baseplate to which are bonded appropriately placed brackets (also of composite) for the support of the optical and detector elements. The focus and orientation of each element is set and maintained by a precision spacer located between the element and its support bracket. The optical system is surrounded by a lightweight rectangular composite enclosure and baffle structure that attaches to and further strengthens the baseplate. Removable covers on the top of the enclosure allow for installation and servicing of the optical and detector elements. An external door mechanism shields the telescope aperture when the instrument is not observing.

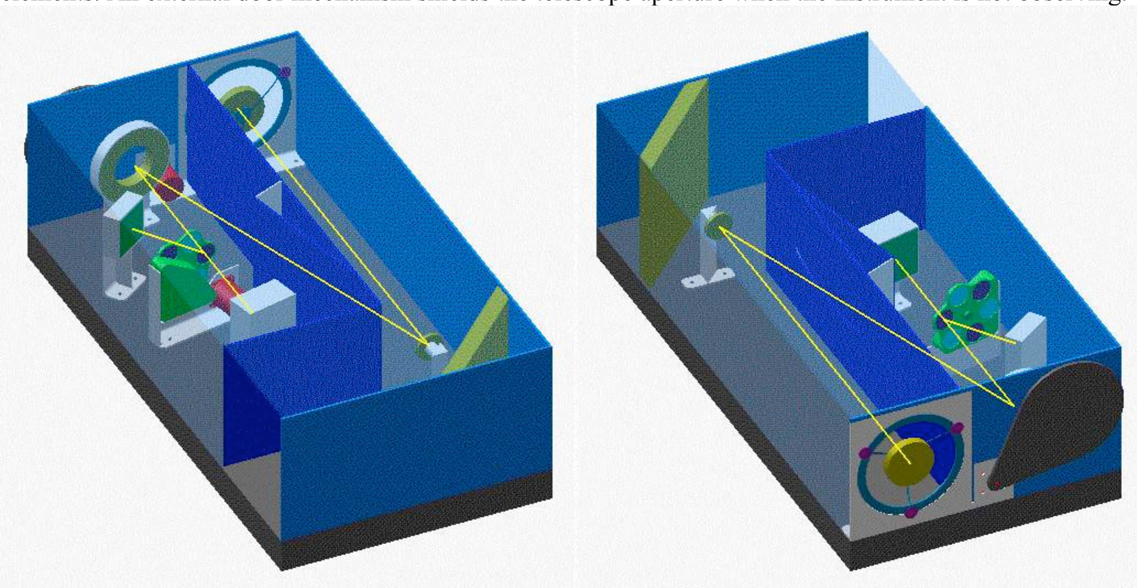


Figure 4.12: Two views of the optical layout of UVC.

Thermal

The thermal control approach is based on shielding the instrument from most direct solar radiation, rejecting light from the solar disc that enters the annular entrance aperture, and an opto-mechanical design that is insensitive to bulk temperature changes. The use of a carbon fibre composite structural system allows the optical system to be almost completely athermalised. Moreover, the use of a low secondary magnification factor in the telescope (i.e. 3:1) makes the optical system less sensitive to dimensional changes in comparison with more traditional space instruments such as TRACE. Preliminary estimates suggest that the allowable temperature range of the opto-mechanical system may be as much as $\pm 50^{o}$ C. Local temperature control will be needed for the liquid crystal polarisation module, which has a smaller allowable temperature range. Thus we consider it prudent to include a small power allocation for thermal control heaters (e.g. 1 W) pending further analysis.

The occulter is the element of the coronagraph, which reaches the highest temperature since it is directly exposed to the Sun, particularly when the Solar Orbiter is near perihelion. The occulter has an exiguous conductive link (represented by the thin rods of the support) with the rest of the structure. The occulter and supporting links are made of titanium and are arranged in a conical geometry such that thermally induced strain is absorbed without de-centring the occulter disc.

The temperature achieved by the occulter has been estimated by assuming a purely radiative heat transfer and an external coating of the same type as that utilised for the sunshield of the payload module. The thermal flux incident on the occulter is 31 kW/m^2 at 0.21 AU and 5.5 kW/m^2 at 0.5 AU. The temperature difference experienced by the occulter along the observation phase is at maximum 266.5 C. A small reduction of the temperature will be caused by the transport of part of the heat through the occulter supports.

Detectors

There are two detectors, optimised for the Visible region (450 – 600 nm) and for the Extreme UV (304-121.6 nm). Both detectors have 15 μ pixels and an array size of 4096 x 4096. The baseline visible detector is an Active Pixel Sensor (APS). This detector system is based on CMOS technology, which is attractive for high radiation dose environment. In addition, the APS architecture has the potential to allow on-chip differencing of the polarimetric signals, yielding substantial improvements in signal-to-noise performance in comparison with conventional CCD detectors.

The baseline EUV detector is a Charge Injection Device (CID) array. This detection system is highly resistant to space radiation. It can easily withstand more than 100 times the lethal dose of a typical CCD and radiation-hardened versions can tolerate doses in excess of 1 MegaRad (Si total dosage). CID detectors have randomly addressable pixels, permitting the dynamically programmable read-out of individual pixels and sub-arrays. This feature will be used in UVC for limiting the readout area to the solar corona (excluding the occulted portion field) and for defining "Regions of Interest" to be observed with higher time resolution. EUV sensitivity will be provided by a Micro-Channel Plate image intensifier coupled to the CID array with fibre-optics. This approach results in a detector that is blind in the visible, substantially improving the stray light rejection performance of the EUV channel.

We note that both detector approaches are based to some extent on anticipated developments in rapidly evolving technologies (e.g. array sizes, APS noise performance). Heavily shielded CCD detectors are suitable alternatives in either case.

4.4.2 Orbit, Operations and Pointing Requirements

4.4.3 Accommodation

4.4.4 Interface and Physical Resource Requirements

Data rates and volume

The primary driver of the data rate is UVC efficiency in the three channels. In the two EUV lines, the coronal signal is weaker and the instrument efficiency lower than in the visible band. Therefore, longer exposure times will be required for the EUV coronal observations. On the basis of the count-rate for H I and He II Lyman-α emission estimated from a coronal hole (i.e. worst case), we assume an average exposure time of 600 sec. A single image (4096 x 4096 pixel) with 16 bit per pixel takes about 270 Mb of memory. Assuming that only 50% of the image is used (the rest is occulted disc, extreme corners of the square matrix, etc.), and that a factor of 3 lossless compression can be achieved, we expect a data rate of 0.07 Mb/s per detector. Since the visible-light detector works simultaneously with the H I Lyman-α (therefore, about 50% of the time), the data volume accumulated per orbit is about 260 Gb. A factor 20 image compression is needed to reach a data rate of about 5 kb/s. It is expected that this level of compression can be achieved with an acceptable loss of information with schemes such as the "Adaptive Discrete Cosine Transform" (ADCT). The ADCT technique is being used in the Large-Angle Spectroscopic Coronagraph (LASCO) on SOHO.

Mass

The UVC mass breakdown is given in Table 4.9.

Component	**Mass (kg)**
Structure	6.6
Thermal control hardware	0.6
Stepper motors	0.26
Optics	2
Mechanisms	1
Detectors (2)	2
CPU & Interface	2
DC/DC converter	1
ADC	1
Data compressor	1
Motor drive	2
Electronic housing	4.2
TOTAL instrument	23.7

Table 4.9: Mass breakdown of the UVC.

Telescope	**Off-axis Gregorian**
Occultation	External occulter: 60 mm diam.
Optics	Primary mirror: 33 mm diam.
	Secondary mirror: 94 mm diam.
	(multilayer coated mirrors)
Eff. focal length	490 mm
Field of View	Annular, Sun-centred. Coverage:
	1.2 – 3.5 R at 0.21 AU,(perihelion at mission start)
	1.7 - 5.0 R at 0.30 AU, (perihelion at mission end)
Spatial resolution	UV (30.4 nm, 121.6 nm): 8 arcsec at 1.2 R (1200 km)
	40 arcsec at 3.5 R (6000 km)
	Visible (500 nm): 40 arcsec entire FOV (6000 km)
Stray-light levels	$< 10^{-8}$ (visible light); < 10-7 (30.4 nm, 121.6 nm)
Wavelength.	1) He II (30.4 ± 2) nm
bandpasses	2) H I (121.6 ± 10) nm
	3) Visible (450 - 600) nm
Detectors	Visible-light: Active Pixel Sensor (APS)
	EUV (30.4 nm., 121.6 nm): Charge Injection Device (CID)
	(EUV intensified); 4096 x 4096 array, 15 μm pixels
Data rate	5 kb/s
Dimensions	80 cm x 30 cm x 30 cm
Mass (kg)	17

Power (W)	25 (operational mode, peak); 20 (stand-by mode)

Table 4.10: UVC instrument summary.

4.4.5 Cleanliness, Ground Operations and Other Requirements

4.4.6 Open Points and Critical Issues

Name / acronym	**Ultraviolet and Visible-light Coronagraph (UVC)**

Objectives	1) To measure the broad-band visible-light polarized brightness (pB) of the K-corona and the narrowband UV Lyman-α line-emissions from coronal hydrogen (λ 121.6 nm) and singly-ionized helium (λ 30.4 nm).
	2) To detect spatial and temporal variations of pB and UV line intensities, and of their ratios.
	3) To provide diagnostics of morphological distribution, density, and kinematics of electrons, hydrogen, and helium in the solar corona.

General description	Coronagraphic imager of the the K-corona pB and of the narrowband UV Lyman-α line-emissions from coronal hydrogen (λ 121.6 nm) and singly-ionized helium (λ 30.4 nm). The design is an externally occulted, off-axis Gregorian. The multilayer-coated mirrors are optimized for λ 30.4 nm, but have still good efficiency at λ 121.6 nm and in the visible. An appropriate filter selects each UV line.

Reference P/L and/or heritage	Modification / improvement of flight H/W based on UVCS/SOHO
	Relevant heritage in the institute 10 years or more experience in the field
	Closest similar instrument developed by other institutes LASCO C1/SOHO

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	CCD camera	Thinned, back-illuminated Marconi
Spectral range	nm	450-600	
Operating T	K	216	
Type	N/A	MCP w. delay-line readout anode	
Spectral range	nm	30 - 1220	
Operating T	K	293	
Optics			
Type	N/A	Reflect.	
FOV	deg.	7	
Bandpasses	nm	30.4 ± 5; 121.6 ± 10; 450 - 600	Al; Al/MgF_2; vis. filters, respectively
Pointing	N/A		
Configuration			
Physical Units	No	1	
Optical Layout	N/A	Off-axis reflecting Gregorian	Externally occulted
Location S/C	N/A		
Physical			
Mass, total	kg	19	(w/o 4.3kg of electr. shielding if shared)
Mass unit 1	kg	19	
Mass unit 2	kg		
Dimension 1	cm	120 x 40 x 30 (L x W x H)	
Power			
Average	W	25	
Peak power	W	25	
Stand-by	W	20	
Data rate / volume			
Average data rate	Bits/sec	72k	Lossless compression (factor of 2)
Peak data rate	Bits/sec	144k	Uncompr. (if compr. by 10 → 20kb/s)
Data volume /orbit	GByte	20	
Own data storage	MByte	--	
Thermal			

Heat load to radiator	W	40	Sun-disk rejection mirror's heat load
Operating T range	K	290 - 300	
Other requirements	N/A		
Cleanliness			
EMC requirements	N/A		
DC magnetic	N/A		
Particulate	N/A	Dust free	Toa avoid light scattering from optics
Molecular	N/A	Organic free	Avoid photo-polymerization on optics
Miscellaneous			
Mechanisms	No.	1 shutter, 1 filter, 1 pointing mec.	
Alignment	Arcmin	< 5 Sun pointing	
Unobstructed FOV	deg	120	
Orbit requirements			
AIT/AIV requirements		TBD	

Development approach / schedule

Preferred model philosophy	EM, QM, FM + refurbished QM
Estimated development time	EM (1 yr), QM (2 yr), FM (1 yr), refurbished QM (1 yr).

Areas considered as critical.

Critical area /unit/ subsystem	Remarks, proposed risk-mitigating measures.
New sensor technology	Development of UV radiation hardened photoemissive detector (e.g., CID)
Multilayer optics with low scatter	Mirror coating technology to be validated via dedicated tests.
Liquid Crystal (LC) polariz. optics	Space qualification, UV radiation and particles survivability

Technology readiness – Design maturity level.

Unit / subsystem	TRL	DML	Justification and remarks.
External occulter	4	5	
Sun-disk rejection mirror	2	5	
LC optics	4	2	

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

4.5 Radiometer (RAD)

4.5.1 Instrument Description

SCIENTIFIC DRIVERS

There is a clear evidence for the influence of the Total Solar Irradiance (TSI) on the terrestrial climate. The unique out-of-the-ecliptic orbit proposed for Solar Orbiter also allows to address most fundamental questions related to the radiative output of the Sun which remain unsolved and cannot be answered by in-ecliptic measurements. It is not known if the solar irradiance variations observed at the Earth are due to solar luminosity variations or to flux redistribution in space. It is also unknown if the radiant output of the poles is different from the values observed from the ecliptic. Such questions can only be answered if observations are carried out from an out-of-ecliptic position. Answer to these questions may also help to explain why the irradiance variations of the Sun are atypically small for its level of activity compared to Sun-like stars and estimate the possible amplitude variations of the solar irradiance over long time-scales.

RAD in itself is able to provide new insight in the origin of TSI variations and to continue the TSI time series. However, the largest scientific output of the instrument would be achieved if direct comparison of its measurements with an in-ecliptic radiometer could be performed. Presently there is a plan that NPOESS on the GOES satellite will also carry out irradiance measurements from the ecliptic at that time.

INSTRUMENT CONCEPT

RAD is a new generation room temperature radiometer. It consists of 4 active cavities which are all pointed towards the Sun. The cavities are operated pair-wise: one cavity (the reference cavity) is heated with a constant electrical power while the second cavity (the measuring cavity) is operated with cyclic shutter phases of typically 100s open and 100s close durations. During practical operations an electronic circuit maintains the heat flux in the measuring cavity constant (and equal to that of the reference cavity) by accordingly controlling the power fed to the cavity heater. All the cavities are capable of making solar irradiance measurements and can be combined with any other cavity.

The primary cavity will yield the continuous observation of the TSI and will be used throughout the mission. The other cavities will be illuminated at different rates, typically once per week and once per month in order to determine the exposure dependent sensitivity changes of the instrument.

The large irradiance changes expected across the orbit and the changes of the solid angle covered by the Sun necessitate a careful instrument design in order to insure highest data quality at all times. Therefore, independent aperture wheels will be mounted in front of each cavity and will have precision apertures of diameter 2mm, 4mm and 8mm in order to keep the energy input into the cavities in a similar range throughout the orbit. The aperture wheels will also be used as shutter mechanisms. The different apertures will be used alternatively depending on the satellite's position in order to keep the level of illumination ”roughly” constant.

The precision apertures will be placed at the front of the instrument, while the view-limiting apertures are placed just in front of the cavities. This arrangement has the advantage to reduce the stray-light and energy input in the radiometer which is specially important at 0.2 AU.

Due to the redundancy of the cavity and aperture arrangements and the possibility to choose any 2 cavities to build a measuring/reference cavity pair, full capability of the instrument is ensured even in the case of an aperture wheel's failure.

Figure 4.13 and 4.14 show an exploded view of the instruments interior and the experiment package respectively, while Figure 4.15 shows its operation principle.

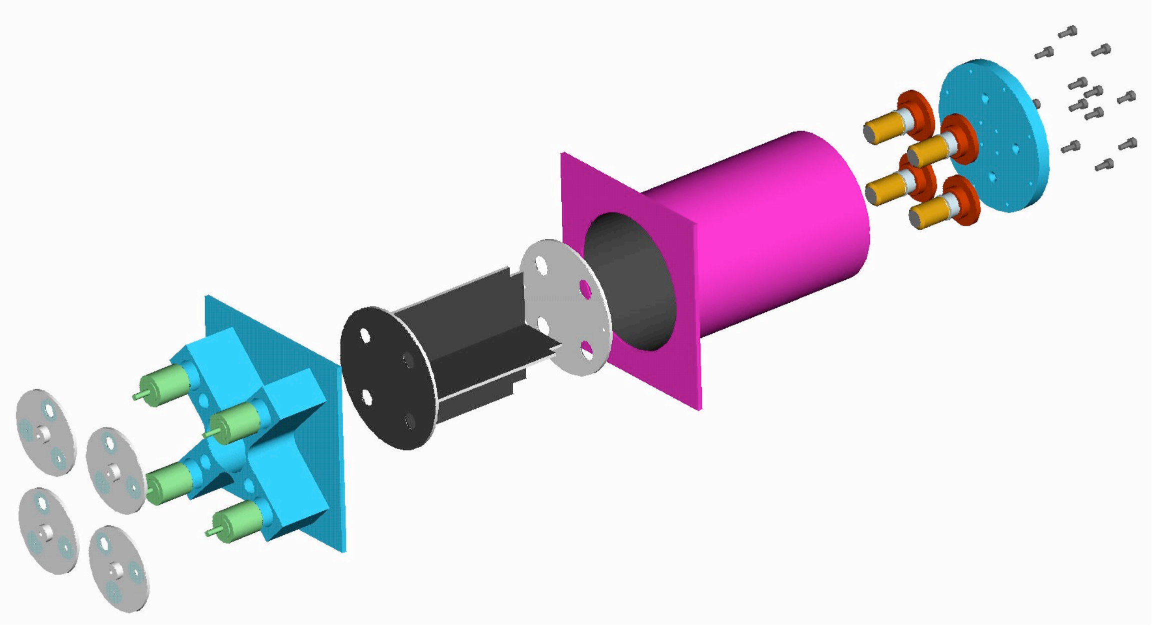


Figure 4.13: Exploded view of the RAD instrument. From front to back: precision aperture wheels (carrying each 3 apertures), motors (green), motor-holder (blue), muffler preventing stray-light between the different channels (black), view-limiting aperture (grey), heat sink (pink) and cavities.

Figure 4.14: View of the RAD instrument package

A front cover common to all four channels will be used only for the launch phase.

The data will be evaluated by a frequency analysis of the instrument's response by extracting the signal at the fundamental of the shutter frequency as was originally proposed for the operation of TIM/SORCE. This operational mode has many advantages over the normal "active cavity" operation in e.g. SOHO/VIRGO, which relies on reaching a steady state before measurements can be taken: for example it reduces the effect of the non-equivalence and removes all the effects taking place at other frequencies.

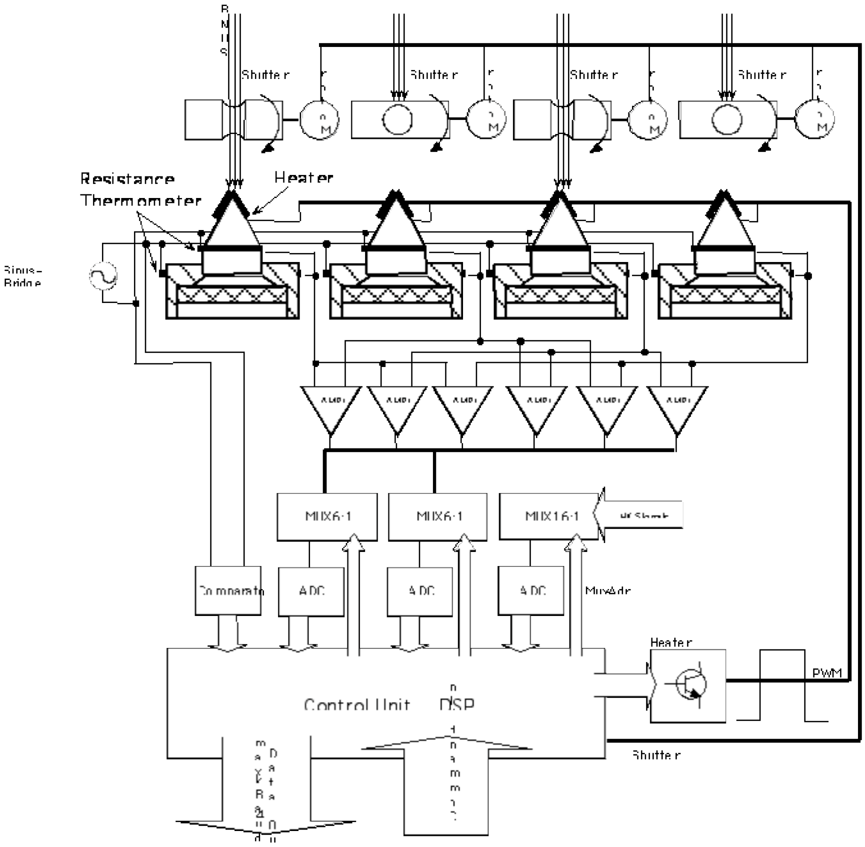


Figure 4.15: Block diagram of the radiometer with the arrangement of the sensor and the control analogue electronics (in this figure, shutter drums are drawn instead of the front aperture wheels).

The electrical power fed to the cavity will be sampled at 100 Hz and will be frequency analysed on-board. The amplitudes and phases of the shutter fundamental frequency and the first few harmonics of it (up to about the 8^{th}) are transmitted to ground. Only the in-phase signal at the shutter fundamental is needed to calculate the irradiance, the harmonics are for the in-flight characterisation of the temporal behaviour of the detectors. Some of the correction factors, needed for the final evaluation of the irradiance, can be determined and/or checked in flight.

The instrument mass breakdown is given in Table 4.11. Note that no sun-shield is included as it was assumed to be provided by the satellite.

Unit	Kg	Material or #prints
Empty package	2.5	Aluminium
Radiometer	2.0	Alu, steel apertures, motors
Electronic	0.7	2 prints
Logic	0.5	1 print
DC/DC	0.9	2 prints
Harness	0.14	0.04 kg + 0.1 kg/m
Total	6.74	

Table 4.11: RAD mass breakdown including 10% contingency

The instrument must have a stable temperature throughout the orbit in order to achi seve its best performances. This is achieved on one hand by keeping the direct radiation entering the instrument approximately constant by the use of different precision apertures and on the other hand by compensating the excess radiation in-falling on the shutters at 0.2AU by heating the instrument with the corresponding difference at farther distances. For this estimation we assume that a front radiation-shield of constant temperature is provided by the spacecraft. Four holes have to be provided in this shield to let the radiation enter the instrument. At 0.2 AU the radiation falling on the instrument shutters/apertures will be larger than at 1 AU. This excess radiation which has to be compensated at large distances from the Sun corresponds to less than 2 W.

4.5.2 Orbit, Operations and Pointing Requirements

RAD will be operated throughout the orbit and will consequently need to be constantly pointed to the Sun. The observing sequences are straightforward since no target selection is necessary. However, the operation of the backup instruments and the change of precision apertures depending on the distance to the Sun have to be monitored. All those commands may be loaded as time-tagged command well in advance for the next orbit. The procedure may change from orbit to orbit depending on the sensitivity changes observed during the preceding orbit and due to the relative behaviour of the primary and backup instruments at intermediate distance, i.e. when changes of apertures need to be performed.

4.5.3 Accommodation

RAD requires 4 holes of 9mm diameter in the front Sun-shield of the satellite and must have an unobstructed field of view of 60°. It has to be pointed to the Sun throughout the orbit and in order to allow proper operation at all times, it should not be located close to parts or instruments, which will undergo large temperature changes.

For the present design, we assumed that a common Sun-shield and global temperature control is provided from the satellite for all the instruments.

4.5.4 Interface and Physical Resource Requirements

The outer dimensions of the RAD instrument (without mounting feet) will be 132 mm x 149 mm x 352 mm. The 132 x 149mm face has to be pointed towards the sun. Note that **no radiation shield** was included in the design since it was assumed that a radiation shield common to all the instruments is provided by the satellite!

In normal operation, RAD's data are reduced on-board and the telemetry rate needed is therefore only 9 bits/s. During test phases, or in case of problems, it will be necessary to transmit the raw measurements which would amount to a data rate of 3.6 kbit/s

RAD has no internal data storage. It was assumed that a central computer would provide this service.

Instrument	Mass (kg)	Power (W)	Volume (cm3)	Data rate (bit/s)
RAD	6.7	7.4	6923	9

Table 4.12: RAD resource requirements.

4.5.5 Cleanliness, Ground Operations and Other Requirements

Cleanliness is not a critical issue for RAD, but it should be built following the same cleanliness specifications as for the other instruments in order to prevent any contamination of their critical optical surfaces.

4.5.6 Open Points and Critical Issues

Memory

RAD has no internal memory to store the measurements. It was assumed that they are stored on a central computer.

Power supply

An option in order to reduce the instrument's weight a common power supply with other instruments would be desirable.

Thermal environment

For this study, it was assumed that a common front radiation shield of constant temperature is provided by the satellite. Since the working principle of RAD is based on a differential temperature measurement, a good thermal stability of the instrument must be insured in order to achieve the desired accuracy throughout the orbit. In particular, it must also be insured that no part of the neighbouring instruments undergo fast temperature changes which would influence RAD's performances. Further refinement of RAD's thermal design will only be possible once the interface to the satellite will clearly be known. (Is a common front radiation shield provided by the satellite? How will the temperature of the surface seen by the instrument vary throughout the orbit?)

Electrical substitution

Due to the large distance variations to the Sun and the solid angle it covers, the area of the cavity which is illuminated and its level of irradiance vary with time. In order to achieve the highest possible performances, the electrical substitution must take place at the same position as where the solar radiation impinges and the level of irradiance has to be kept within ranges. The level of irradiance is taken into account by the use of the different precision apertures. The dimensions of the instruments were chosen so as to optimise the position of the cavity illumination at the extremes of the orbit and it shows its feasibility. However, during the development phase a theoretical investigation should be carried out in order to characterize the performances of the instrument at the intermediate positions and determine if they could be significantly improved by the use of multiple and independent heating circuits in the cavities. Since the apertures of the different cavities are totally independent, it will even be possible to monitor eventual performance changes of the primary instrument at the times of the aperture changes by measuring simultaneously with one of the backup cavities. This will also provide an independent test of the non-equivalence between the solar and electrical substitution.

Name / acronym	**Solar Absolute Radiometer / RAD**

Objectives	1) to measure solar irradiance
	2) to measure solar irradiance variations to an accuracy of 0.01%
	3) to provide diagnostic tools for understanding solar irradicance variations
	4) to compare polar from equator irradiance

General description	Differential solar absolute radiometer. The design is based on active radiometers using a phase-sensitive detection scheme.

Reference P/L and/or heritage	Improvement of flight H/W based on VIRGO
	Relevant heritage in the institute: 20 years or more experience in the field
	Data analysis scheme as proposed by the TIM/SORCE team

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Active cavity	
Spectral range	Nm/eV	Solar spectrum	
Operating T	K	Room temperature	
Optics / antennas			
Type	N/A	Absorbing	
FOV	Deg	4.5	Unobstructed view from the front part within 60° full angle to avoid stray light in the instrument
Bandpass	Nm/eV	N/A	
Pointing	N/A	Satellite pointing is ok	
Configuration			
Physical Units	No	1	
Unit layout	N/A		
S/C related requirements	N/A	DPU, time base, thermal control	Front radiation shield and temperature control needs to be provided by the satellite. 4 holes of 9mm diameter in the front radiation shield are necessary.
Physical			
Mass, total	kg	6.7	
Mass unit 1	kg	6.7	
Mass unit 2	kg		
Dimension 1	cm	13 x 15 x 35	Without mounting feet
Dimension 2	cm		
Power			
Average	W	7.4 (=5.14+2.24)	5.4 W: standard, +2.24 W at 1 AU
Peak power	W	TBD	
Stand-by	W	Non-op heaters 12W	Not really applicable, since the instrument is supposed to measure all the time, including the cruising phase.
Data rate / volume			
Average data rate	Bit/sec	9	
Peak data rate	Kbit/sec	3.6	During check-out time or in case of problems
Data volume /orbit	MByte	15	
Own data storage	MByte	None	

Thermal			
Heat load to radiator	W	12	
Operating T range	K	280-320	
Other requirements	N/A	No fast T drifts of nearby parts/instruments	
Cleanliness			
EMC requirements	N/A	N/A	
DC magnetic	N/A	N/A	
Particulate	N/A	N/A	
Molecular	N/A	N/A	
Miscellaneous			
Mechanisms	No.	4 shutter/aperture wheels 1 front cover wheel	
Alignment	Arcmin	<10 Sun pointing	
Unobstructed FOV	deg	60 (full angle)	
Orbit requirements		Pointed to the Sun at all times!	
AIT/AIV requirements		Clean room class 100000	Or as required by other instruments

Development approach / schedule

Preferred model philosophy	EM, PFM
Estimated development time	Radiometer (1 yr), EM (1 yr), P FM (1 yr)

Critical areas: Technology readiness – Design maturity level

Critical area /unit/ subsystem	TRL	DML	Justification and remarks
Active Cavity	9	3	Flown on VIRGO
Thermal control	9	2	Flown on VIRGO need to be adapted for Solar orbiter
Lock-in electronics	2	5	Need to be space qualified

Technology Readiness Level (TRL):
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

5. Additional Instruments

5.1 Spectrometer/Telescope for Imaging X-rays (STIX)

5.1.1 Instrument Description

STIX is an x-ray imaging spectrometer, operating from 3 to 150 KeV, which determines the location of x-ray emission from the Sun as a function of time and energy.

The primary scientific objective of the STIX instrument is to establish the timing, location and spectra of energetic electrons near the Sun. This will enable these electrons to be related to subsequent observations by the in-situ solar energetic particle and radio instruments. In this way, **STIX serves as a high-energy link between imaging and in-situ observations.**

Secondary science objectives are threefold:

- To determine the size and morphology of hot thermal and nonthermal x-ray sources with subarcsecond resolution, which is not feasible at a few keV with such instrumentation at 1 AU because of diffraction.
- To use comparisons with observations from 1 AU or other solar-orbiting spacecraft (if available) to measure the directivity of solar x-ray emission. This will provide the first direct measurements of beaming at the Sun.
- To use observations of 'over-the-limb' flares (in conjunction with other spacecraft, if available) to isolate the weak coronal component of hard-x-ray bursts, so as to characterize the chromospheric-coronal transport of energetic electrons. (Because of the difference in coronal/chromospheric intensities, it is difficult to do this with conventional imaging.)

Imaging technique
STIX imaging uses the same indirect imaging technique as used on Yohkoh / HXT. Imaging information is encoded in the relative count rates in separate detector elements located behind pairs of grids, each of which absorbs a distinct directionally-sensitive fraction of the incident flux.

The telescope hardware employs a set of 64 subcollimators, each of which consists of a pair of widely separated, x-ray opaque grids with an x-ray detector element located behind the rear grid. (See Figure 5.1) Front and rear grid pairs have identical pitch and orientation, whose choice determines the spatial frequency to be measured. As was demonstrated by Yohkoh/HXT, the relative count rates of a pair of subcollimators, one of whose grids is displaced by _ of its pitch, can be used to accurately measure both the real and imaginary parts of one Fourier component of the angular distribution of the source. With 64 subcollimators, the imaging system then measures 32 different Fourier components. This data can then be used to reconstruct the source image, using well-established techniques used by radio astronomy, Yohkoh/HXT and RHESSI.

Since the detector spatial resolution need match only the ~cm dimension of the subcollimator, detectors can be optimized for their spectral resolution. This enables the imaging to be done as a function of energy, resulting in an instrument which functions as an imaging spectrometer. Such a detector system consists of a set of four identical CdZnTl solid-state detector modules each of which is divided into sixteen 1 cm square elements. Operating at room temperature, they provide a total of 64 detector elements, each of which provides 2 to 4 keV resolution from 3 to 150 keV. Large arrays of CdZnTl detectors will be flown next year on NASA's SWIFT mission.

The front and rear grid assemblies are each constructed as a unit by stacking sets of etched tungsten sheets up to a thickness of 1 mm. Grid pitch values range from 35 microns to 2 mm. Grids with such parameters are less demanding than those successfully fabricated for RHESSI. STIX-compatible prototype grid assemblies are being fabricated at present by Mikro Systems Incorporated using NASA SBIR funding. At a separation of 1.4 m, the corresponding spatial resolution of the individual subcollimators (defined as one half of the ratio of grid pitch to separation) is 2.5 arcseconds to 5 arcminutes. At 0.2 AU, however, the former is equivalent to 0.5 arcseconds at 1 AU.

The FWHM **imaging field of view**, given by the ratio of the subcollimator diameter (1cm) to separation is 24 arcminutes, which even at 0.2 AU is sufficient to fully encompass an active region. In addition, STIX also can do spatially-integrated spectroscopy over a much wider field of view (3 degrees). Such a field is sufficient for observing events anywhere on the Sun, even from 0.2 AU. Furthermore, within this wider field of view STIX can determine the centroid of the source location to an accuracy of arcminute or better. This 3-degree **locating field of view** enables every transient x-ray event to be associated with its active region of origin and so provides **a capability that could provide useful input for target selection.** The overall effective area of the system for imaging is about 16 cm^2, which at 0.2 AU is equivalent to more than 4 times that of RHESSI.

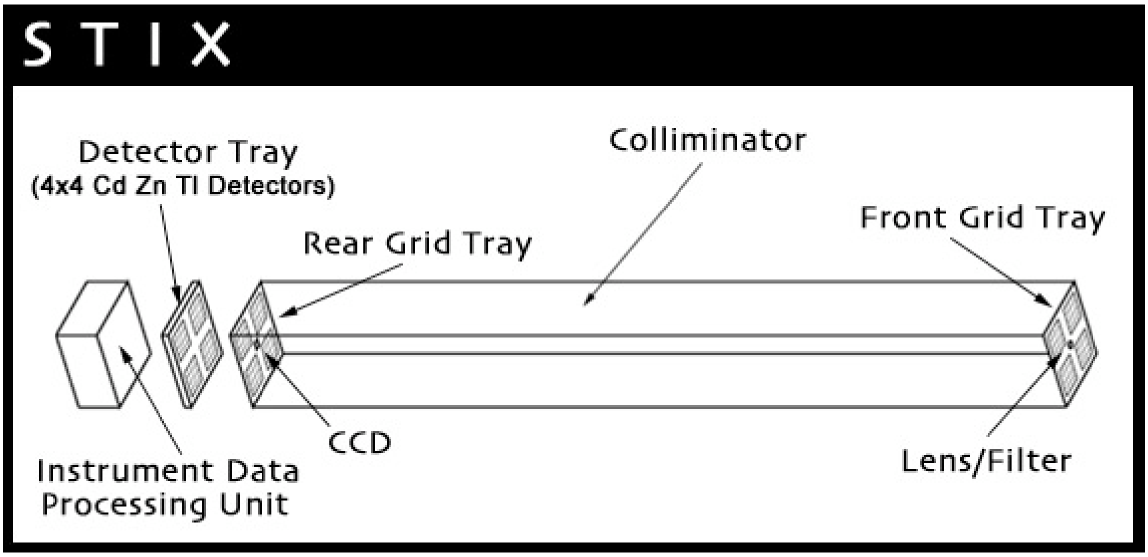


Figure 5.1: STIX configuration. A conceptual illustration of the STIX instrument **that does not show the current aspect system or sunshade**, both of which are described below.
Overall dimensions are 12 x 12 x 150 cm.

Data handling

Expected count rates from the detector system will vary from a few counts/second during background periods to more than 10^6 counts/second during intense flares. The key on-board data handling challenge is to process and compress this data in order to allow ground-based image reconstruction from a modest telemetry volume. No image reconstruction is done on board.

Each detected photon generates an output pulse from a single CZT detector element. Such analog pulses are shaped and amplified by front-end electronics and then digitized into one of 16 energy channels. Initial data processing consists of accumulating such events according to their energy and detector into one of 16x64 (1024) energy/detector bins. A basic instrument time resolution of 1/8 second results in an initial data rate of ~16 kBytes/second. A rotating 64-Mbyte buffer stores ~1 hour of this full-resolution data within the instrument.

Within the context of this 1-hour time frame, an autonomous instrument processor is used to form detector- and time-averaged spectra and detector-and energy-averaged light curves. Enhanced count rates in the light curves are used to identify flare time intervals for imaging. The processor then calculates statistically significant sums over adjacent time bins and/or energy channels. The data for a single image is then in form of 64 2-byte numbers, representing the counts in each detector element for the selected time/energy interval. The image morphology and location is represented by the *relative* values of these counts and can be expressed as a corresponding set of 64 4-bit binary fractions relative to the maximum count among the 64 values. Compressing the total counts to 8 bits, the image can then be stored as this 8-bit total plus 64x4 bits of relative counts plus 7 bytes of miscellaneous information for a total of 40 bytes per image.

Assuming a long-term average of 6 minutes of flare data per hour, imaging in an average of 10 energies bands with an average 2-second cadence implies a requirement of **1800 images per hour**. Adding 25% for aspect, housekeeping and non-imaging datasets results in **an average data rate of 200 bits/second**.

5.1.2 Orbit, Operations and Pointing Requirements

Elements of a pair of limb-sensing aspect systems are embedded into the grids. For each aspect subsystem, the front grid contains a Fresnel lens element, whose openings are restricted to a narrow band in 1-dimension. This focuses a one-dimensional profile of the Sun onto the rear grid plane, where a wavelength-filtered linear APS detector, mounted on the rear grid plane, determines the effective limb position in 1-dimension with ~1 arcsecond precision. An identical, but orthogonal subsystem determines the limb position in the other dimension. The combination of these two systems determines the pitch/yaw offset of the STIX imaging axis with respect to Sun center. The concept is an adaptation of the aspect system on RHESSI which determines the solar aspect to ~0.4 arcsec rms at up to a 128 Hz cadence. For STIX, the cadency requirement could be significantly eased. T**he output of this system could be made available in real-time to provide ~arcsecond relative aspect for use by other instruments.** STIX will rely on S/C systems to provide absolute roll aspect to ~ 3 arcminutes.

The use of Fresnel lens elements eliminates the possibility of degradation of optical surfaces and **restricts the internal solar heat load to a few 10's of milliwatts**. As demonstrated by RHESSI, the use of an embedded aspect system significantly eases alignment and pointing requirements. This enables the use of a relatively lightweight metering structure between the front and rear grid planes. The primary alignment requirement is to maintain the relative twist of the front and rear grid assemblies to about 3 arcminutes, which is about x20 less stringent than on RHESSI. Thus STIX instrument provides **arcsecond-class imaging with arcminute-class alignment requirements**. In effect it substitutes aspect knowledge for mechanical control. As described, the system will determine the absolute location of x-ray sources to 1 arcsecond within a 24 arcminute field of view.

STIX should be Sun-pointed and co-aligned with the other imaging instruments to ~2 arcminutes to provide imaging overlap. Operations would be autonomous, based on preloaded parameter settings. Examples of such parameters include gain-setting parameters (to match gains of the individual detector elements) and parameters used by the instrument data processor algorithm for the selection of imaging intervals.

The science output of the STIX instrument would greatly enhanced by observations during the cruise phase. The telemetry requirements during this period could be tailored by selecting only large flares for analysis.

5.1.3 Accommodation

The thermal shielding issue for STIX is readily address by an opaque sunshade. The entrance aperture of STIX can be covered by a sunshade in the form of C-C in front of multilayer Beryllium. This is not a problem for STIX imaging because it requires shielding in any case to suppress the intense flare flux of low energy solar x-rays. Although not thermally significant, this low energy x-ray flux would otherwise overwhelm the detectors' ability to handle individual photons. For a 3 keV observing threshold, the optimum sunshade thickness for x-rays is equivalent to ~1 mm of Carbon or 3 mm of Beryllium. With suitable design, such a sunshade should provide suitable thermal protection as well.

The sunshade would require 2 open apertures (each ~0.1 x 50 mm) in front of the aspect elements. At 0.2 AU such apertures would transmit about 340 mw. About 50 mw of this would be transmitted to the lower grid tray inside the instrument while the remainder would be absorbed or reflected by the top tungsten grid. (If necessary a thin reflective coating could be applied to the grids without affecting their imaging performance.) One unresolved issues are whether the sunshade should be provided by the s/c or by the instrument. An additional design issue is the tradeoff between co-alignment accuracy of the sunshade apertures (that determines their width and transmitted heat load) and the feasible level of heat dissipation/reflection by the front grid.

5.1.4 Interface and Physical Resource Requirements

The estimated mass of STIX is 4 kg. This is based on a well-calculated mass of 0.25 kg for the grids, 0.2 kg for the sunshade and 1 kg each for the metering structure, detectors and electronics. An additional 25% (1 kg) is added for contingency to reach the value tabulated below.

5.1.5 Cleanliness, Ground Operations and Other Requirements

5.1.6 Open Points and Critical Issues

The primary technical issues to be resolved are related to the aspect system. Suitable sensors need to be identified and evaluated for radiation hardness. In addition, ground tests of a prototype aspect system should be undertaken to provide early validation of its optical design.

Name / acronym	**Spectrometer/Telescope for Imaging X-rays (STIX)**

Objectives	
	1) to establish the timing, location and spectra of energetic electrons near the Sun and so provide a high-energy link between in-situ and imaging observations.
	2) to determine the size, and morphology of hot thermal and nonthermal x-ray sources with subarcond resolution, (not achievable from 1 AU due to diffraction)
	3) in conjunction with other spacecraft, to measure the directivity (beaming), and the chromospheric/coronal transport of energetic electrons,

General description	An x-ray imaging spectrometer which uses indirect (Fourier) imaging to achieve 2.5 arcsecond imaging with high spectral resolution from 3 to 150 keV.

Reference P/L and/or heritage	Yohkoh / HXT
	RHESSI

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	4 4x4 arrays of 1cm x 1cm CdZnTl detectors	
Spectral range	KeV	3 – 150 keV	
Operating T	C	-25 to +25	Not critical
		Linear CCDs or photodiode arrays for embedded aspect system	
Optics / antennas			
Type	N/A	Tungsten x-ray grids	
FOV	sr.	24 x 24 arc minutes for imaging. 3x3 degrees for spectroscopy and source location.	~0.25 g/cm^2 low-Z window required to suppress low energy x-rays. Also addresses thermal issues.
Bandpass	N/A	3 – 150 keV	Responsive to all energies.
Pointing	N/A	Solar-pointed	Maintains roll alignment to 3 arcmin relative to s/c roll sensor.
Configuration			
Physical Units	No.	Single box	OR Detector/electronics could be mounted separately with ~ 1 mm alignment to a grid optics module
Layout	Cm	12 x 12 cm side faces Sun	
Location S/C	N/A	Sun-pointed, but prefer shadowing on the other 5 sides.	
Physical			
Mass, total	kg	~5	Includes 1 kg contingency
Dimensions	cm	~12 x 12 x 150	Long axis may be cylindrical with a slightly larger diameter.
Volume	liter	~20	
Power			
Average	W	~ 4	
Peak power	W	~ 4	Average is same as peak.
Stand-by	W	~ 1	TBD
Data rate / volume			
Average data rate	Bits/sec	200	
Peak data rate	Bits/sec	~40000	
Data volume /orbit	MByte	320 for 149 day orbital period	Data selection and internal storage can maintain average data rate

Own data storage	MByte	~ 64 megabytes	
Thermal			
Heat load to radiator	W	None to radiator	
Operating T range	C	-20 to +25	Not critical.
Other requirements	N/A	None special	
Cleanliness			
EMC requirements	N/A	None special	
DC magnetic	N/A	None special	
Particulate	N/A	None special	
Miscellaneous			
Mechanisms	No.	None	No deployables or expendables.
Alignment		Not critical	
Orbit requirements		None special	
AIT/AIV requirements		May require radioactive source for testing/calibration	No purge required.
Alignment		Relies on s/c to provide roll aspect to 3 arcminutes	
Sunshade		C-C + Be sunshade in front of front surface	

Development approach / schedule

Preferred model philosophy	Provide EM and FM. Project determines need for other models.
Estimated development time	EM (2 yr), FM (1 yr)

Areas considered as critical

Critical area/unit/subsystem	**Remarks, proposed risk-mitigating measures.**
Aspect subsystems	Need to identify suitable radiation-hard linear CCD or photodiode array for aspect subsystems.

Technology readiness – Design maturity level

Unit/ subsystem	**TRL**	**DML**	**Justification and remarks**
Detectors	7	3-5	Adaptation of current designs. Detector type is space-qualified.
Grids	9	2	Prototype grids, designed for STIX are being fabricated at present.
Aspect subsystem	9	6	Concept used successfully on RHESSI. Major modifications needed to optical design, which can be validated through ground tests of a bread-board version.

<u>Technology Readiness Level (TRL)</u>:
1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

<u>Design Maturity Level (DML)</u>:
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

5.2 Heliospheric imager (HI)

5.2.1 Instrument Description

A heliospheric imager extends the view of the heliosphere from ~5 solar radii (at 0.2 AU) to as much as the anti-solar direction. Thus, it would have almost the full Earth-Sun line within its field of view for major parts of the observation time, depending on the spacecraft attitude. This instrument would allow us for the first time to observe the propagation of transient disturbances towards the Earth from various distances and aspect angles, in particular from outside the ecliptic plane. The combination of the Heliospheric Imager and the Ultraviolet, Visible light Coronagraph provide a complete picture of the solar corona from an out of ecliptic perspective. The strawman instrument is similar to that being constructed for the Solar Terrestrial Relations Observatory spacecraft. The instrument consists of a large external baffle equipped with a simple white light telescope. The baffle design has sufficient margin to accommodate the instrument off pointing and the variable size of the sun. If more complete spatial coverage of the solar disk is required, two HI units will be necessary, one for each hemisphere. Preliminary design studies have demonstrated that each unit can probably be built for less than 5 kg as shown in Table 5.1. If the structure and door could be shared within another instrument, the mass would be reduced. A crucial accommodation requirement would be that the hemisphere to be watched must be kept free of Sun-illuminated spacecraft appendages.

Table 5.1: HI strawman mass breakdown

Unit (1/2 HI)	Mass (kg)
One telescope	1.29
Camera electronics box	0.5
Structure	1.8
Kinematics structure mounts	0.3
Baffle cover assembly (door)	0.754
Contamination components	0.150
Alignment cube	0.013
Margin	0.2
Total	5.007

5.2.2 Orbit, Operations and Pointing Requirements

Orbit
Being largely sheltered behind the heat shield, the HI has no directly solar illuminated critical surfaces. Care must be taken to isolate the first solar illuminated baffle edge from the rest of the instrument. The varying angular size of the sun as well spacecraft pointing variation are accommodated by designing the baffles for a worst case situation. This worst case design will necessarily obscure a modest portion of the inner corona.

Operations
We expect that HI will obtain a single summed frame every hour. This cadence will provide adequate coverage of the outer corona on temporal scales similar to the C3 coronagraph on SOHO. An externally provided, imaging processing computer will remove cosmic rays from the images, sum the images and compress the combined frame.

Pointing

The HI optical axis should be nominally held to within 5 arc-minutes of the payload optical axis. The pointing of the HI optical axis should be held to <1 arc-minute stability over a 2 minute exposure. The HI should be sufficiently well baffled that the payload optical axis can be pointed within 1.5 degrees of sun center. It is believed that these pointing requirements will not drive the design of the solar orbiter.

5.2.3 Accommodation

The HI must be mounted on the side of the spacecraft with a clear view of about a hemisphere. Small protrusions into this hemisphere are allowable but their stray light properties must be understood. The structure supporting the baffle/telescope/camera must be mounted with minimal mechanical distortion to the spacecraft.

5.2.4 Interface and Physical Resource Requirements

Diagram TBD.

Estimates of the physical resource requirements are given in Table 5.2 for each optical sensor. The sequencing and image processing electronics is not included in this estimate.

Table 5.2: HI strawman physical resources (single sensor)

Parameter	Resource
Mass	5kg
Power	Average: 1W, Heater power: 3W
Envelope	75x30x15cm
Telemetry	1.6kbps after summing and compression

5.2.5 Cleanliness, Ground Operations and Other Requirements

Cleanliness
The instrument external surface must be kept clean to 300A per Mil 1246B. The instrument will be double bagged and purged during AIV. A sacrificial layer of MLI may be desirable. When the instrument is opened during spacecraft AIV, a locally clean area must be arranged. We anticipated that the on shot door will be fired twice during AIV.

5.2.6 Open Points and Critical Issues

A critical enabling technology for HI is the active pixel sensor based camera. The envisioned camera simplifies the instrument. Since the APS is not as susceptible to radiation damage, the required degree of cooling is reduced. The APS detector also does not require a shutter. A nominal APS camera for HI is expected to be 2048x2048 with 14 bit digitization and a net >200kHz readout rate possibly through multiple ports.

Name / acronym	Heliospheric Imager/HI

Objectives	
	1) directly observe velocities and morphology of coronal mass ejections at large solar radii from an out of ecliptic perspective.
	2) directly view longitudinal streamer magnetic configurations and the coronal large scale structures at great heliocentric distances.
	3) characterize the mass transport and energy balance of the outer corona.

General description	Two forward looking white light telescopes sheltered from stray radiation by a baffle enclosure. Ideally, one telescope should be located for a polar view and another telescope for a streamer view.

Reference P/L and/or heritage	
	e.g. employs elements of HI design developed for STEREO.
	e.g. Relevant heritage dating back to the 1960s
	e.g. builds on extensive coronagraph heritage in the US and Europe.

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	2	Active pixel sensor	
Spectral range	nm	400-1100	
Operating T	C	-50 degrees	Preliminary based on radiation damage calculations performed for the solar probe mission.
format		2048x2048	
Optics / antennas			
Type	assorted	Glass lenses	
FOV	deg	>30	The instrument should have as large a field of view as is practical within a single telescope.
Bandpass	Nm	500-700	Preliminary value only
Pointing	Minutes of arc	<1 over 2 minutes	
Configuration			
Physical Units	2		
Unit layout	TBD		Follows HI.
S/C related requirements		Field of regard of each telescope covers roughly a hemisphere. Protrusions would not necessarily be a problem but would need to be modelled for stray light. Kinematic mount desired.	
Physical			
Mass, total	kg	10	Preliminary estimate based on HI design. Does not include cables, blankets and/or electronics box.
Mass unit 1	kg	5	
Mass unit 2	kg	5	
Dimensions	cm	Nominally 75x30x15	Long dimension is toward the sun. Based on reasonable scaling of HI design.
Power			
Average electronic	W	1	Will probably need some additional heater power. Assumes spacecraft

			provided sequencing and power conditioning.
Peak power	W	Nominally 2	
Stand-by	W	0.2	
Heater power	W	3 (per unit)	
Data rate / volume			
Average data rate	Bits/sec	1600	One frame from each telescope with a 10:1 lossy compression per each hour. Image summing and compression by the spacecraft computer is assumed
Peak data rate	Bits/sec	TBD	
Data volume /orbit	KByte	TBD	
Own data storage	MByte	None planned	
Thermal			
Heat load to radiator	W	TBD	APS radiator needs a reasonably clear field of view to deep space.
Operating T range	K	TBD	
Other requirements	N/A	Does not look directly at the sun.	
Cleanliness			
EMC requirements	N/A		
DC magnetic	N/A		
Particulate	1246C	External surfaces 200-300	Preliminary value
Molecular	1246C	External surfaces A (100Angstroms of nonvolatile residue)	Preliminary value
Miscellaneous			
Mechanisms	No.	None	
Alignment	Arcmin	<60 Sun pointing	
Unobstructed field of regard	deg	Desired to have a clear hemisphere for each telescope, some protrusions are allowed.	
Orbit requirements		Out of ecliptic view needed	
AIT/AIV requirements		TBD	

Development approach / schedule

Preferred model philosophy	EM, QM, FM
Estimated development time	EM (1 yr), QM (2 yr), FM (1 yr).

Critical areas: Technology readiness – Design maturity level

Critical area /unit/ subsystem	TRL	DML	Justification and remarks
Active Pixel Sensor	2	6	We are just at the beginning of understanding these devices for scientific space applications. A CCD camera could be substituted but there would be a resource penalty.
Optics/baffle	5	3	The design would be similar to that used for the HI instrument on STEREO. HI has undergone bread/board and component level validation in preparation for the STEREO SECCHI PDR.

Technology Readiness Level (TRL):

1: basic principles observed and reported; 2: technology concept and application formulated; 3: analytical and experimental critical function, characteristic proof-of-concept; 4: components validated in the laboratory; 5: component and/or breadboard validation in a relevant environment; 6: system demonstrated in relevant environment (ground or space); 7: system prototype validated in space environment; 8: system flight-qualified through tests; 9: system verified by successful mission.

Design Maturity Level (DML):
1: existing HW; 2: existing + minor modifications; 3: existing + major modifications, 4: new, detail design available; 5: new, preliminary design available, 6: new, conceptual design available.

5.3 Gamma-ray detector (GRD)

5.3.1 INSTRUMENT DESCRIPTION

The Gamma-ray Detector will most likely consist of a high-Z scintillator such as NaI or CsI, viewed by a photomultiplier (PMT). It needs to be surrounded by an anticoincidence shield (probably a plastic scintillator) to eliminate penetrating particle background. Possibly the same scintillator could be used for neutrons by using pulse shape discrimination (PSD) to separate gamma-rays from neutrons. It may even be possible to use this scintillator for total energy measurement in a dE/dx vs E telescope for the Energetic Particle Detector (EPD).

A high voltage power supply (HVPS) is needed for the PMT. The PMT signal would require standard analog preamp/amp/discriminator followed by an analog-to- digital converter (ADC). The anticoincidence shield might require a separate PMT with its own analog electronics chain. Possibly a single PMT could be used with phoswitch circuitry to discriminate between the plastic and CsI or NaI pulses. The resulting digital data would be accumulated into a memory and read out by the DPU.

5.3.2 ORBIT, OPERATIONS AND POINTING REQUIREMENTS

No orbit requirements but best measurements close to Sun. Continuous operation preferred. No pointing requirements.

5.3.3 ACCOMMODATION

The sunward side of detector should not have much high-Z material in front (Al,Mg,C, etc.).

5.3.4 INTERFACE AND PHYSICAL RESOURCE REQUIREMENTS

Volume depends on packaging and sharing, but detector, PMT, HVPS and preamp is 10 cm dia. x 15-20 cm. Also one electronics board ~100-200 cm^2 area.

Bit rate ~200 bps average, ~1 megabytes storage desirable since prime data is in bursts.

5.3.5 CLEANLINESS, GROUND OPERATIONS AND OTHER REQUIREMENTS

No special requirements.

5.3.6 OPEN POINTS AND CRITICAL ISSUES

Main issue is exploring combining with other instruments.

Name / acronym	**Gamma-Ray Detector (GRD)**

Objectives	1) to provide measurements of solar gamma-ray lines and continuum.
	2) to help elucidate solar eruptive events (e.g. CMEs, flares) and understand the sun as a prolific and variable particle accelerator.
	3) to provide information on elemental abundances in both accelerated particles and the solar atmosphere.

General description	A non-imaging detector measuring gamma-ray spectrum in the energy range from ~0.3 to ~10 MeV. A number of different sensor types may be capable of performing the desired measurements.

Reference P/L and/or heritage	Gamma-ray detectors on CGRO, SMM, Yohkoh, RHESSI, etc.

Parameter	Units	Value / Description	Remarks
Sensor / detector			
Type	N/A	Scintillator or semiconductor device. Also need PMT or PD for anti-coincidence shield.	A number of different technologies may be proposed.
Spectral range	MeV	~0.3 to 10	
Operating T	C	-20 to +30 C	Not critical
Optics / antennas			
Type	N/A		
FOV	sr.	4□	<3 g/cm^2 obscuration in the nadir direction.
Bandpass	N/A	N/A	Responsive to all energies.
Pointing	N/A	On nadir (+X) face of spacecraft	Details not critical.
Configuration			
Physical Units	No.	Single box	
Layout	Cm	~15 x 15 x 15	Size will depend on sensor type.
Location S/C	N/A	On shadowed equipment deck (+X). Would prefer location on shadowed boom, if available.	Prefer away from spacecraft to minimize background, but not critical.
Physical			
Mass, total	kg	2 to 3.5 for standalone unit	Mass will depend on sensor type and degree of subsystem sharing.
Dimensions	cm	~15 x 15 x 15	Size will depend on sensor type and degree of subsystem sharing. Cubic shape not likely.
Volume	liter	3 to 5	See above.
Power			
Average	W	1.5 to 3.0	Power will depend on sensor type and degree of subsystem sharing.
Peak power	W	1.5 to 3.0	Average is same as peak.
Stand-by	W	1.5 to 3.0	Some sensor types may have reduced power in stand-by, others not.
Data rate / volume			
Average data rate	Bits/sec	<160	
Peak data rate	Bits/sec	~8000	During solar flares (≤ 1000 sec).
Data volume /orbit	MByte	<250 for 149 day orbital period	Data rate constant over orbit.
Own data storage	MByte	~ 2 megabytes	Negotiable with spacecraft.

Thermal			
Heat load to radiator	W	None to radiator	
Operating T range	C	-20 to +30	Not critical.
Other requirements	N/A	None special	
Cleanliness			
EMC requirements	N/A	None special	
DC magnetic	N/A	No strong B-fields near unit	B-field could affect PMT, if used.
Particulate	N/A	None special	
Miscellaneous			
Mechanisms	No.	None	No deployables or expendables.
Alignment		Not critical	
Orbit requirements		None special	Nadir (+X) to Sun.
AIT/AIV requirements		May require radioactive source for testing/calibration	No purge required.

Development approach / schedule

Preferred model philosophy	Provide EM and FM. Project determines need for other models.
Estimated development time	EM (1.5 yr), FM (1 yr)

Areas considered as critical

Critical area/unit/subsystem	Remarks, proposed risk-mitigating measures.
None identified.	All pertinent technologies fairly well in hand. Modeling of specific sensors required.

Technology readiness – Design maturity level

Unit/ subsystem	TRL	DML	Justification and remarks
ND		5	Detailed designs could be generated fairly quickly for most sensor types that may be proposed.
ND	7-9		The detector technologies are fairly mature and have been flown in space in various incarnations but the specific systems that will be proposed for the SO have not flown and will be new designs.

Technology Readiness Level (TRL):
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